CPTR 70
April 2001

OVERVIEW OF PULSE DETONATION PROPULSION TECHNOLOGY

M. L. Coleman

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REPORT DOCUMENTATION PAGE	Form Approved OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave Blank)*	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	April 2001	Technical Report, 1942 to 1999

4. TITLE AND SUBTITLE

Overview of Pulse Detonation Propulsion Technology

5. FUNDING NUMBERS

C:SP0700-97-D-4004

6. AUTHOR(S)

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7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

The Johns Hopkins University
Chemical Propulsion Information Agency
10630 Little Patuxent Parkway, Suite 202
Columbia, MD 21044-3204

8. PERFORMING ORGANIZATION REPORT NUMBER

CPTR 70

9. SPONSORING/MONITORING AGENCY NAME (S) AND ADDRESS(ES)

DTIC - AI
8725 John J. Kingman Road
Suite 0944
Ft. Belvoir, VA 22060-6218

Naval Air Warfare Center Weapons Division
Code 477000D
China Lake, CA 93555-6100

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

11. SUPPLEMENTARY NOTES

Hardcopies available from CPIA only. CPIA's DTIC-assigned source code is 423900.

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Approved for public release; distribution is unlimited.

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

Propulsion systems based on the pulsed detonation cycle offer the potential to provide increased performance while simultaneously reducing engine weight, cost, and complexity, relative to conventional propulsion systems currently in service. The increased performance potential is due to the high thermodynamic efficiency of detonation combustion. The near constant volume heat addition process of the detonation cycle, along with the lack of a compression cycle lend to the high efficiency and specific impulse, simplicity, and low-cost of pulse detonation propulsion systems. Pulse detonation Engines (PDEs) have the potential for operation at speeds ranging from static to high-supersonic (M<5), with competitive efficiencies enabling supersonic operation beyond conventional gas turbine engine technology. Currently, no single engine cycle exists that has such a broad range of operability. Pulse Detonation Rocket Engines (PDREs) have the potential to drastically reduce the cost of upper stage and orbit-transfer propulsion systems, and are also attractive for lunar and planetary exploration, and landers and excursion vehicles that require throttling for soft landing.

This report discusses the thermodynamic basis of pulse detonation propulsion technology and identifies the major technology initiatives currently underway in the United States.

14. SUBJECT TERMS

PULSE DETONATIONS, DETONATIVE COMBUSTION, PDRE (PULSE DETONATION ROCKET ENGINES), PDE (PULSE DETONATION ENGINES), THRUST, INTEGRATED SYSTEMS, TACTICAL WEAPONS, PROPULSION SYSTEMS, SUPERSONIC FLIGHT, ROCKET ENGINES, AIR BREATHING ENGINES

15. NUMBER OF PAGES

68

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
UNCLASSIFIED	UNCLASSIFIED	UNCLASSIFIED	UL

NSN 7540-01-280-5500

Standard From 298 (Rev. 2-89)
Prescribed by ANSI Std. Z39-18
298-102

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The Government Administrative Manager for CPIA is the Defense Technical Information Center (DTIC), Code DTIC-AI, 8725 John J. Kingman Road, Suite 0944, Ft. Belvoir, Viriginia 22060-6218. The Government Technical Manager (Contracting Officer's Technical Representative) is Mr. Stuart Blashill, Naval Air Warfare Center Weapons Division, Code 477000D, China Lake, California 93555-6100.

PREFACE

This Chemical Propulsion Technology Review continues CPIA's recurrent series of technical summaries and status reports on topics pertaining to missile, space, and gun propulsion technology. The general aim is to collect, analyze, and discuss technology advancements in a language understood by a broad range of propulsion technologists.

This CPTR presents a technical review of pulse detonation engine and pulse detonation rocket engine science and technology. Deflagration and detonation combustion processes are compared briefly, detonation combustion physics are reviewed, benefits of detonation combustion at the systems level are discussed, and current technology efforts are presented.

CPIA solicits comments on the technology review effort, including suggestions on topics for future issues. For technical comments or suggestions, contact Mr. Tom Moore, CPIA Technical Services Supervisor, at 410-992-9951, ext. 207, or Mr. Mark Coleman at 410-992-9950, ext. 210. Individuals employed by organizations that subscribe to CPIA services may request personal copies of this document by contacting CPIA at 410-992-7300, cpia@jhu.edu, or http://www.cpia.jhu.

ABSTRACT

Propulsion systems based on the pulsed detonation cycle offer the potential to provide increased performance while simultaneously reducing engine weight, cost, and complexity, relative to conventional propulsion systems currently in service. These improvements can be traced to the high thermodynamic efficiency of the nearly constant-volume combustion cycle and the low entropy rise in the working fluid produced by detonation. The pulse detonation cycle can be applied to both airbreathing and rocket-based systems, and pulse detonation engines may require less packaging volume than conventional propulsion systems due to their inherent simplicity. In addition, airbreathing pulse detonation engines can potentially operate over a wide range of flight Mach numbers ($M=0$ to 5). These characteristics combine to make pulse detonation propulsion systems potentially attractive to a wide range of military and commercial missions. Recent advancements in measurement, diagnostic, and control technologies coupled with advancements in computational combustion dynamics and computers have created the environment where development of practical pulse detonation propulsion systems may now be now possible.

The near constant-volume heat addition process of the detonation cycle, along with the lack of a compression cycle lend to the theoretical high efficiency and specific impulse, simplicity, and low-cost potential of pulse detonation propulsion systems. Pulse Detonation Engines (PDEs) have the potential to operate statically and accelerate from low subsonic through high supersonic velocities, with competitive efficiencies enabling supersonic operation beyond conventional gas turbine engine technology. Currently, no single engine cycle exists that has the ability to operate over such a broad range of flight velocity (Mach 0 to 5). Pulse Detonation Rocket Engines (PDREs) have the potential to drastically reduce the cost of upper stage and orbit-transfer vehicle propulsion systems, and are also attractive for lunar and planetary exploration vehicles, planetary landers and excursion vehicles that require throttling for soft landing, and space vehicle attitude control systems.

Development of practical PDEs and PDREs will introduce many new component, subsystem, and system-level design challenges. Practical systems will require development of fast acting, flightweight propellant valves, advanced combustion control systems, efficient inlets and nozzles, and system specific component integration design solutions. In addition, operational systems must be designed to operate with practical fuels and propellant combinations, such as JP-10/air, RP-1/O_2, and H_2/O_2.

This report reviews the conventional Chapman-Jouguet detonation theory; conceptual airbreathing and rocket-based pulse detonation propulsion system designs; and the goals and objectives of technology development programs currently underway in the United States.

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GLOSSARY

AF	Air Force
AFB	Air Force Base
AFRL	Air Force Research Laboratory
AFRL/PR	AFRL Propulsion Directorate
AFRL/PRSA	AFRL Propulsion Sciences and Advanced Concepts Division
AFRL/PRST	AFRL Turbine Engine Division Combustion Sciences Branch
APRI	Advanced projects Research Incorporated
APS	Airbreathing Propulsion Subcommittee
ASI	Adroit Systems Incorporated
ASTP	Advanced Space Transportation Program
CFD	Computational Fluid Dynamics
C-J	Chapman-Jouget
CT	California Institute of Technology
DARPA/TTO	Defense Advanced Research Projects Agency/Tactical Technology Office
DDT	Deflagration to Detonation Transition
DFRC	Dryden Flight Research Center
DoD	Department of Defense
GE	General Electric
GEAE	General Electric Aircraft Engine
GH_2	Gaseous Hydrogen
GRC	Glenn Research Center
IRAD	Independent Research And Development
IVTAN	Institute of High Temperature, Russian Academy of Sciences
LaRC	Langley Research Center
LMTAS	Lockheed Martin Tactical Aircraft Systems
Lox	Liquid Oxygen
MEM	Micro Electro-mechanical
MSFC	Marshall Space Flight Center
MURI	Multidisciplinary University Research Initiative
MVLSO	M. V. Lomonosov State University
NASA	National Aeronautics and Space Administration
NAWC	Naval Air Warfare Center
NPS	Naval Postgraduate School
NRL	Naval Research Laboratory
ONR	Office of Naval Research
PDE	Pulse Detonation Engine
PDET	Pulse Detonation Engine Technologies
PDRE	Pulse Detonation Rocket Engine
PSU	Pennsylvania State University
RBCC	Rocket-Based Combined Cycle
RevCon	Revolutionary Concepts in Aeronautics program
RPAV	Revolutionary Propulsion for Aeronautical Vehicles
SAIC	Science Application International Corporation
SBIR	Small Business Innovative Research
SU	Stanford University
TRL	Technology Readiness Level
UCSD	University of California San Diego
UF	University of Florida
US	United States
UTRC	United Technologies Research Corporation
UTSI	University of Tullahoma Space Institute
ZND	Zel'dovich, von Neumann & Doring

NOMENCLATURE

c	Sonic Velocity
c_p	Constant Pressure Heat Capacity
c_v	Constant Volume Heat Capacity
γ	Gamma, Ratio of Specific Heats (c_p/c_v)
h	Enthalpy
H_2	Hydrogen
K	Coefficient of Thermal Diffusivity
kcal	Kilocalorie
M	Mach Number
m/s	Meters Per Second
Mole	6.022×10^{23} Molecules in a Mole
msec	Millisecond
O_2	Oxygen
P	Pressure
ΔP	Change in Pressure
P_o	Initial Pressure
P_1	Gas Pressure Immediately Behind Detonation Combustion Wave
q	Heat Added to System
ρ	Density
R	Specific Gas Constant
τ	Reaction Rate
T	Temperature
T_b	Temperature of Burned Gas
T_o	Initial Temperature
T_u	Temperature of Unburned Gas
T_1	Temperature of Gas Immediately Behind Detonation Combustion Wave
u	Reactant Velocity
V_D	Chapman-Jouget Detonation Velocity of the Combustible Gas Mixture
V_{det}	Detonation Velocity

1.0 INTRODUCTION

Pulse Detonation Engines (PDEs) and Pulse Detonation Rocket Engines (PDREs) detonate combustible propellant mixtures to produce high chamber pressures and thrust. Practical PDE and PDRE designs may include multiple detonation chambers to obtain high aggregate operating frequencies and quasi-steady thrust. Current combustion and system models predict very high propulsion efficiencies for PDE and PDRE devices and good thrust characteristics from the low subsonic to the high supersonic flight regimes. Potential performance advantages over constant-pressure combustion devices include lower specific fuel consumption, higher specific impulse, and higher thrust-to-weight characteristics.

Detonations enable very rapid material and energy conversion. This rapid material conversion rate, or burning rate, does not allow enough time for the local expansion of the combustion products to occur. Therefore, the detonation process is thermodynamically closer to a constant volume process than the constant pressure process typical of conventional deflagration-based propulsion systems. The higher thermodynamic efficiency of the nearly constant volume combustion process (detonation) is directly traceable to the lower entropy rise in the working fluid, when compared to the constant pressure (deflagration) combustion process. The high-pressure ratios associated with detonation combustion may eliminate the need for expensive, high-pressure feed systems, thereby reducing propulsion system weight, complexity, cost, and packaging volume. In addition, the pulsed detonation cycle can operate over a wide range of flight Mach numbers without the assistance of booster stages, and can be applied to a wide range of military, civil, and commercial missions and systems.

Pulse detonation propulsion technology has received considerable attention in the United States (U.S.) over the past decade. Renewed interest in pulsed detonation for propulsion applications is due to the potential for substantial improvement in performance, and substantial reductions in propulsion system weight, complexity, cost and specific fuel consumption, relative to propulsion systems currently in service.

Practical application of pulsed detonation for propulsion requires the ability to couple an increase in thermal efficiency to an increase in propulsion efficiency. Before detonations can be used effectively in a controlled manner for propulsion applications, the physics behind reliably establishing a detonation and its propagation along a combustor axis must be understood. Numerous agencies within the U.S. Government, academia, and industry are currently pursuing initiatives to characterize fundamental detonation physics for select propellant combinations, demonstrate single and multi-cycle detonations, develop critical components for prototype systems, and acquire test data to validate performance models.

This report provides an overview of combustion physics and historical detonation research, describes the thermodynamic basis for PDE and PDRE performance, and summarizes the PDE and PDRE technology development efforts currently underway in the U.S. In addition, this report reviews PDE and PDRE engine cycle operation and reviews conceptual PDE and PDRE system designs. Specific engineering issues and technology areas requiring further study are also highlighted.

2.0 DEFINITIONS RELATED TO COMBUSTION PHENOMENA

Established usage of certain terms related to combustion phenomena can be misleading. Before proceeding with the topic of pulsed detonation propulsion technology it is useful to review applicable terminology.

2.1 COMBUSTION

Combustion is defined here as an exothermic chemical reaction between a fuel and an oxidizer that once initiated can sustain itself as long as the ingredients are present in the proper proportions and thermal diffusion limits are not exceeded. However, not all combustion events are the same. The velocity at which a combustion wave propagates through a propellant mixture is an accurate measure of the strength or violence of the event. Combustion wave velocity is dependent on several factors including mixture composition, pressure, temperature, and the geometry of the volume where the combustion occurs. In general, a combustion wave is considered a deflagration, although the detonation wave is another class of the combustion wave[1].

There are two mechanisms whereby energy required for activation of a chemical reaction can be transferred from the reacted material to the unreacted material — thermal radiation and diffusion and shock pressure forces (mechanical shock and compression). Thermal radiation and diffusion are the mechanisms that propagate chemical reactions in deflagrations and explosions. Material surrounding an initial chemical reaction is warmed above its decomposition temperature to sustain the reaction. Mechanical shock and compression are the mechanisms that initiate chemical reactions in detonations. Compression forces imposed on unreacted material by a supersonic detonation wave causes rapid heating and subsequent combustion of the reactants to sustain the reaction [1,2].

2.2 DEFLAGRATION

Deflagration is the common combustion phenomena associated with current flight propulsion systems such as ramjets, turbojets, and rockets. A deflagration is a rapid chemical reaction in which the heat output is sufficient to enable the reaction to proceed and be accelerated without input of heat from another source. Deflagration is a surface phenomenon with the reaction products flowing away from the unreacted material along the surface at a subsonic velocity. Thermal energy release is the mechanism that sustains the reaction[1,2,3]. The deflagration flame speed is a function of pressure, temperature, and turbulence of reactants, and the permissible range is correctly predicted by many classic laminar and turbulent flame theories[4,5]. The deflagration flame speed is dependent on the chemical composition, mass diffusion rates, and thermal transfer rates of the reactants. Typical flame speeds for a deflagration combustion wave are 1 to 30 m/s[6,7]. The effect of a deflagration under confinement is an explosion. Confinement of the reaction increases pressure, rate of reaction, and temperature, and may cause transition to detonation[3]. Though deflagration is the most common combustion process, it is theoretically not the most efficient thermodynamic path for combustion to occur because the entropy of the resulting gases is maximized, which reduces the amount of energy available to do useful work.

2.3 CHEMICAL EXPLOSION

A chemical explosion is a chemical reaction or change of state that is effected in a very short period of time and generates a large volume of high temperature gas. The exothermic reaction rate increases exponentially with the subsequent increase in temperature and pressure. An explosion produces a shock wave in the surrounding medium. Even though the explosion is powerful and occurs very fast, the combustion event itself occurs as a deflagration wave as it travels through the unburned reactants. The thermal energy released during the reaction sustains the reaction. In a deflagration the combustion reaction process and shock wave propagation process proceed in an uncoupled manner[1,2,5,6,7].

2.4 DETONATION

A detonation is a violent chemical reaction that proceeds through the reacted material toward the unreacted material at supersonic velocity. The supersonic combustion event propagates at high velocities and produces a rapid and violent combustion of the reactants due to the strong shock wave leading the detonation. In a detonation the combustion reaction and shock wave propagation proceed in a totally coupled and mutually supporting manner. The shock imposed on the unreacted material by the supersonic combustion wave causes a rapid heating and subsequent combustion of the reactants to sustain the reaction. The reaction is sustained through the propagation of the reaction coupled to the shock wave, and is described by the Chapman-Jouguet theory discussed in the following sections[1, 2, 3, 4, 5, 7].

3.0 COMBUSTION OVERVIEW

3.1 DEFLAGRATION WAVE STRUCTURE

The classic method used to analyze combustion waves is a long tube filled with a combustible mixture of gases. An ignition source is used to initiate combustion. Figurès 1 and 2 provide a comparison of steady state deflagration and detonation combustion wave properties as the combustion waves propagate relative to the reactants. In Fig. 1, mass flow of the combustible gas mixture is flowing from left to right towards a deflagration combustion zone. Figure 1 shows the variation of gas temperature and concentration of reactants across the combustion zone.

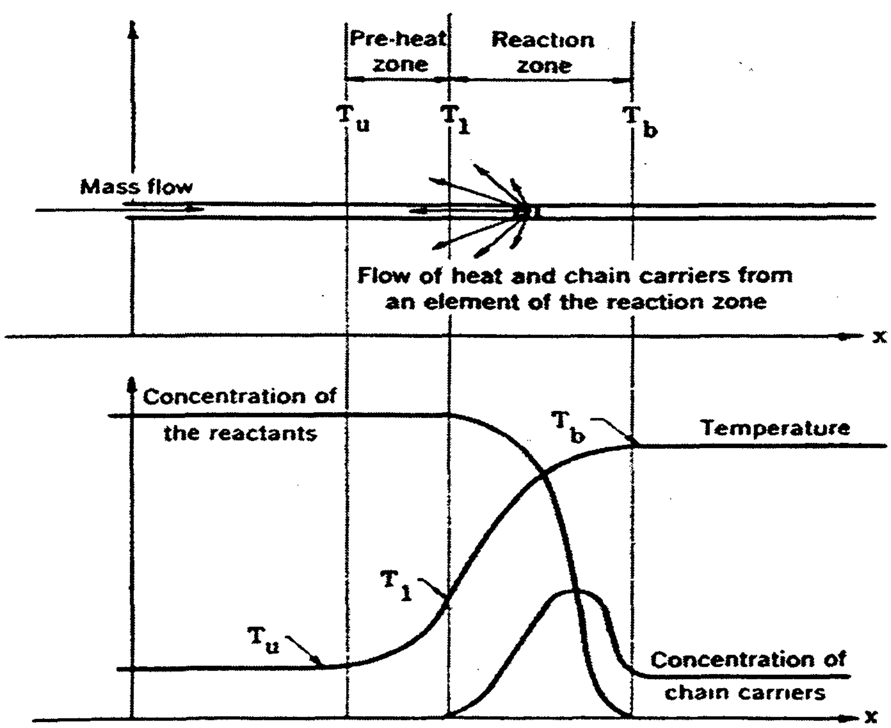

Figure 1. Propagation and Structure of a Deflagration Combustion Wave in a Tube Filled with Combustible Gas Mixture [5]

Between the boundaries u and b a chemical reaction occurs and molecules of reactants diffuse in the direction u to b, and molecules of the products of combustion diffuse in the direction b to u. T_u is the temperature of the unburned gas and T_b is the temperature of the burned gas. As can be

seen from the figure, the deflagration flame front gradually raises the temperature of the unburned gas from T_u to T_1 before the onset of the chemical reaction. The rate of rise of temperature for one-dimensional heat flow is given by $K\partial^2T/\partial x^2$, where K is the coefficient of thermal diffusivity. The positive value of the second derivative between T_u and T_1 indicates that the gas mixture receives by conduction more heat from the hotter gas downstream than it loses to the cooler gas upstream. T_1 marks the inflection point in the curve beyond which the second derivative is negative, indicating that after T_1 the gas loses more heat to the upstream gas than it receives from the downstream gas[5].

The pressure varies slightly across the deflagration flame front shown in Fig. 1 due to the confinement of the tube. If the deflagration were occurring in the open atmosphere, the pressure would equalize immediately and the pressure of the reactants would equal the pressure of the combustion products. The confinement imposed on the deflagration flame front by the tube in Fig. 1 results in a slight expansion, reducing the pressure of the combustion products.

3.2 DETONATION WAVE STRUCTURE

Detonation combustion wave properties are shown in Fig. 2 for a detonation propagating in a tube filled with a combustible gas mixture. The tube is closed at one end and open at the other end. The detonation is initiated near the closed end of the tube and is propagating toward the open end.

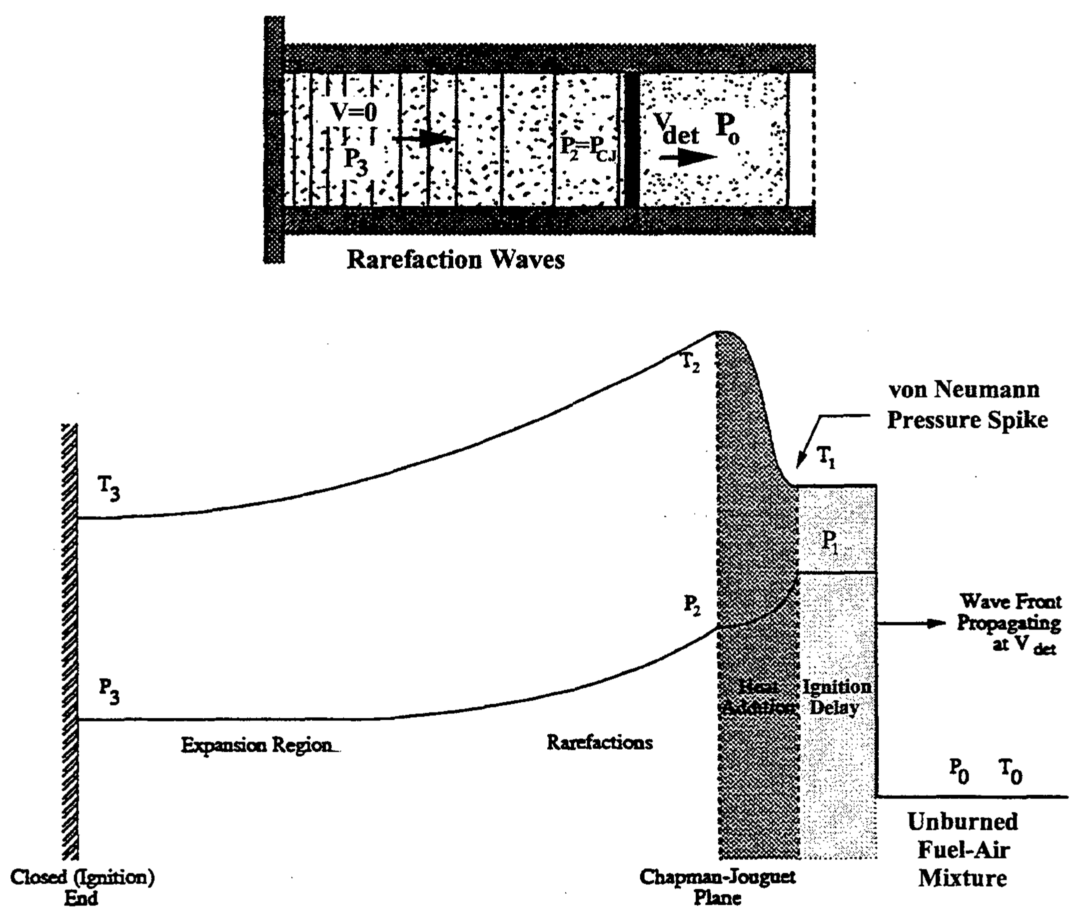

Figure 2. Propagation and Structure of a Detonation Combustion Wave in a Tube Filled With Combustible Gas Mixture [8]

The detonation wave can be modeled as a strong shock that rapidly compresses the reactants to initiate combustion, and a thin flame front in which heat addition occurs. The shock front moves at the detonation velocity, V_{det}, relative to the gas and dramatically increases the temperature and pressure of the gas from initial values T_o and P_o to T_1 and P_1. The region of unburned gas immediately behind the shock is a stable high-pressure region known as the von Neumann spike. This region represents the ignition delay, and its width is dictated by chemical kinetics of the gas mixture. Once the chemical reaction is initiated heat is added to the flow causing the temperature to increase and the pressure to decrease. The width of the heat addition region is determined by the time to complete the combustion reactions[8]. At this point the burned gas is at state 2, which corresponds to Chapman–Jouguet conditions for a self-sustaining detonation. The temperature, pressure, and density of thė gas at state 2 are significantly greater than at state 1.

The pressure and density in the stable detonation wave (P_2, ρ_2) are significantly lower than in the von Neumann region between the shock front and the chemical reaction zone. However, the detonation wave temperature (T_2) just behind the flame region is significantly higher than in the von Neumann spike. In closed tube detonations, an expansion region exists behind the heat addition region. Rarefaction waves emanate from the closed end to ensure that the normal velocity of the gas at the wall is zero. As a result of the expansion, most of the burned gas in the detonation tube is at pressure P_3, which is significantly lower than the pressure just behind the detonation wave[8].

In a detonation the combustion reaction and shock wave propagate in a coupled and mutually supporting manner. Zel'dovich (1940)[9], von Neumann (1942)[10], and Doring (1943)[11] believed the detonation wave could be viewed as three distinct regions whose widths are dependent on the equivalence ratios and kinetics of the gas mixture in which the detonation wave is propagating. Figure 3 shows the thermodynamic properties in the regions of the commonly named ZND detonation wave structure.

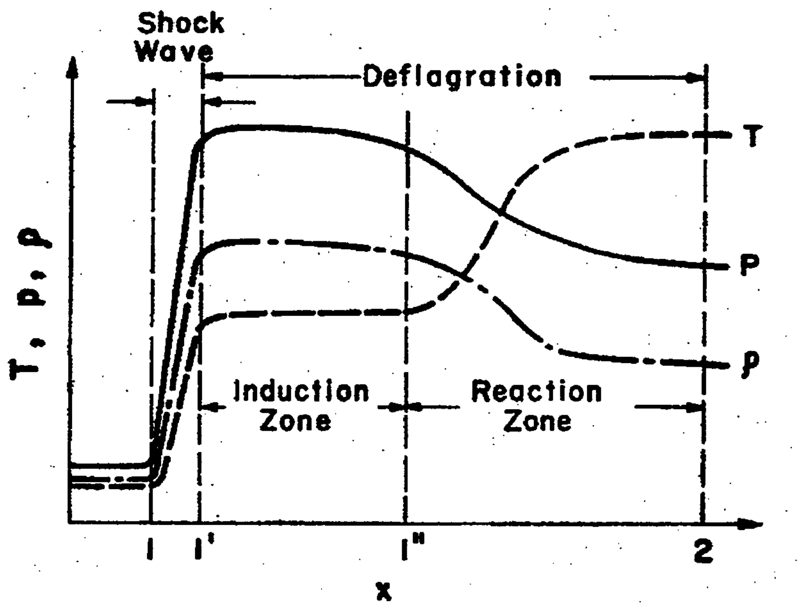

Figure 3. **Physical Properties of the 1-D Detonation Wave structure** [4A]

The first region, the shock wave, has a width of just a few Angstroms, yet delivers a tremendous amount of energy into the unburned reactants. This energy input results in immediate and dramatic increases in pressure, density, and temperature. The dramatic increase in the thermodynamic properties of the gas mixture increases the chemical reaction rates and accelerates the energy release phase of the wave structure. The deflagration region consists of two zones that describe

the thermodynamics of combusting the reactants. The first, which is known as the Induction zone, has a relatively short width in which the chemical reaction is beginning but is not yet impacting the thermodynamic properties of the gas mixture. The Induction zone transitions to the Reaction zone when the reaction rate begins to increase exponentially, driving temperatures up and stabilizing pressure and density to their final equilibrium value. The total width of the three zones is on the order of one centimeter, and each zone is dependent on the next to sustain the detonation wave[4, 7].

Although the one-dimensional ZND model has worked well for approximating detonation wave structure, in actuality the detonation wave has a complex three-dimensional structure. The three-dimensional structure is the result of transverse shock waves that propagate laterally behind the leading normal shock wave. The intersection of the transverse waves with the leading normal shock wave results in localized high-pressure, high-temperature regions known as triple points. The extreme high heating that occurs at these points greatly accelerates the local reaction rates and ensures that the heat release region is closely coupled to the leading normal shock wave. The rapid oscillation of the triple points across the leading shock wave promotes the stability of the detonation wave and results in the characteristic "fish scale" patterns commonly seen in soot foil traces. Soot foil traces are typically obtained by placing thin sheets of tin or stainless steel treated with fine soot particles into recessed areas inside of experimental detonation tubes. The depth of the recess inside the detonation tube is equal to the thickness of the soot-treated metal so that the detonation wave sees a constant-area cross-section as it traverses the length of the tube. Detonation pressure imbeds the soot into the tin or stainless steel "foil" leaving a "footprint" of the detonation wave structure as the detonation wave traverses the tube. An example of a hydrogen-air detonation wave structure obtained using the soot foil trace technique is provided in Fig. 4.

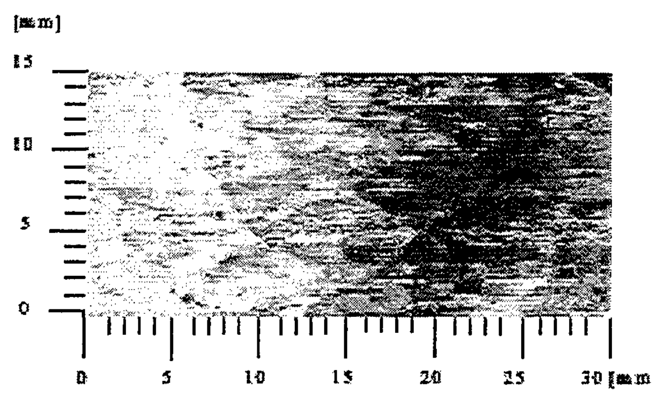

Figure 4. Hydrogen-Air Detonation Wave Structure Soot Foil Trace

A qualitative comparison between a deflagration and a detonation is given in Table 1 for gaseous fuels and oxidizers [7]. The reference frame for this analysis is the one-dimensional stationary combustion wave shown in Fig. 5. The properties used to describe the event include ratios of reactant velocity (u), sonic velocity (c), density (ρ), temperature (T), and pressure (p). Subscripts denote the burned and unburned side of the combustion wave. The results shown in Table 1 depict the dramatic differences between deflagration and detonation combustion events.

TABLE 1. Qualitative Difference Between Detonation and Deflagration in Gaseous Fuel/Oxidizer Mixture [7]

Reacted/Unreacted Property Ratios		Deflagration	Detonation
Wave Velocity Ratio	u_1/c_1	0.0001-0.03	5 - 10
Reactant Velocity Ratio	u_2/u_1	4 – 6 (acceleration)	0.4-0.7 (deceleration)
Pressure Ratio	p_2/p_1	~ 0.98 (slight expansion)	13 – 55 (compression)
Temperature Ratio	T_2/T_1	4 – 16 (heat addition)	8 – 21 (heat addition)
Density Ratio	ρ_2/ρ_1	0.06 – 0.25	1.7 – 2.6 (higher compression)
Relative Reaction Rate Ratio	τ_1/τ_2	1	~ 200

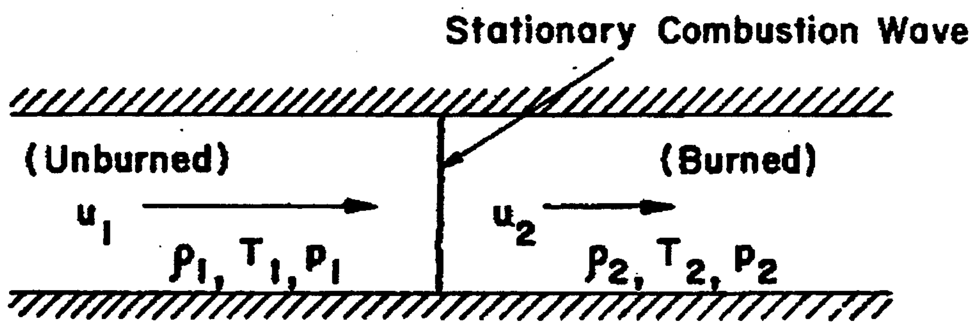

Figure 5. Schematic of Stationary 1-D Combustion Wave [R-06]

4.0 HISTORICAL OVERVIEW

4.1 CHAPMAN-JOUGET THEORY

The first recognized detonation was discovered and later patented by A. Nobel in 1864. Nobel and his father invented a mercury fulminate ignitor that initiated a detonation in a nitroglycerine charge, and later perfected the process with the invention of dynamite [12]. In the 1870's, other researchers began to relate the strength of an explosion to how it was initiated and its propagation velocity. At this point in time leading researchers hypothesized that detonations were initiated by some form of mechanical shock, and that the shock was the mechanism that sustained the detonation. By 1880, researchers concluded from numerous tests of different fuels and oxidizers at different equivalence ratios that detonation velocity is uniform and only dependent on the fuel and its mixture ratios [13]. In 1883 researchers demonstrated that, under the right conditions, deflagrations would transition into a detonation wave. This new discovery led to experimental proof that the detonation process can be viewed as a rapid adiabatic reaction whose energy release drives the detonation wave, and the supposition that there exists a inherent relation between the chemistry of the reactants, the conditions in which they are ignited, and the detonation properties they exhibit [13].

Early discoveries and ideas prompted continued research and analysis, and by 1890 researchers were able to show the detonation pressure as a function of detonation velocity and the reactants heat of reaction. Using the Rankine theory, V. A. Michelson was able to show that there are two possible solutions for combustion. Michelson was the first to conclude that reactants at different conditions will burn naturally around two distinct conditions. He also noted that there was a convergence of pressures at the upper point, correlating with the detonation process [14].

The combined work of D. L. Chapman and E. Jouguet confirmed the work of early researchers while working independently during the late 1890's and early 1900's. Publishing in 1899, Chapman stated that there exists a minimum velocity in which a detonation can occur, and it is thermodynamically tied to the properties of the burned gas [15]. Jouguet worked from 1901-1905 and established the relation that the detonation wave velocity is equal to the sound velocity of the burned gas in which it propagates [16]. He verified this result by comparing computed results with the experimental results of several of his predecessors. J. L. Crussard validated the Chapman-Jouguet (C-J) theory in 1907 by relating the two specific combustion pressure points on the Hugoniot curve (pressure-specific volume adiabat) [17]. C-J theory is recognized as the relationship between velocities of combustion wave processes and the pressures at which they occur. C-J theory postulates that there are two regions at which combustion process can occur.

Assuming steady, one-dimensional flow in the constant-area combustor shown in Fig. 5, with no external heat added or rejected, negligible interdiffusion effects, and no viscous effects, the Hugoniot relationship can be derived from the conservation equations. By analyzing the detonation in this form, it can be viewed as a supersonic shock wave with calculable properties in front of and behind the wave.

The governing equations of a thermodynamic process can be derived from the basic conservation equations:

Continuity equation:

$$\frac{d(\rho\mu)}{dx} = 0 \tag{1}$$

Conservation of momentum:

$$\rho\mu\frac{du}{dx} + \mu\frac{d(\rho\mu)}{dx} = -\frac{dp}{dx} \tag{2}$$

Conservation of energy:

$$\rho\mu\left[\frac{d}{dx}\left(h + \frac{\mu^2}{2}\right)\right] = -\frac{d}{dx}\left(q_{cond}\right) \tag{3}$$

Where enthalpy (h) and heat added (q) to the system are defined by:

$$h = CpT + h^0 \tag{4}$$

$$q = h_1^0 - h_2^0 \tag{5}$$

$$q = -\lambda\frac{dT}{dx} \tag{6}$$

The Hugoniot curve is defined by the relationship between enthalpy (h), pressure (p), and density (ρ) of the gases in a combustion event. The relationship between h, p, and ρ is directly related to the various combustion conditions. The fourth and final equation for deriving the Hugoniot relation is based on the assumption that the gases in both the burned and unburned regions behave like a perfect gas. The perfect gas law is defined for both regions, where R is the specific gas constant for the reactants.

$$\text{Perfect gas law:} \qquad p = \rho RT \tag{7}$$

With integration, substitution and manipulation of intermediate equations, the following equations can be derived:

$$\frac{\gamma}{\gamma-1}\left(\frac{p_2}{\rho_2}-\frac{p_1}{\rho_1}\right)-\frac{1}{2}(p_2-p_1)\left(\frac{1}{\rho_1}+\frac{1}{\rho_2}\right)=q \tag{8}$$

$$h_2-h_1=\frac{1}{2}(p_2-p_1)\left(\frac{1}{\rho_1}+\frac{1}{\rho_2}\right) \tag{9}$$

Equations (8) and (9) are forms of the Rankine-relation, but are formally named for Hugoniot who derived them and fit the various combustion conditions to the pressure vs. specific volume ($1/\rho$) curve. Equation (8) provides the relation between heat addition and the gases initial and final pressures and densities.

The Hugoniot curve shown in Fig. 6 describes the different conditions at which combustion can occur. These combustion conditions include various strengths of deflagrations and detonations, dependent upon the pressure and specific volume conditions at which the event is occurring.

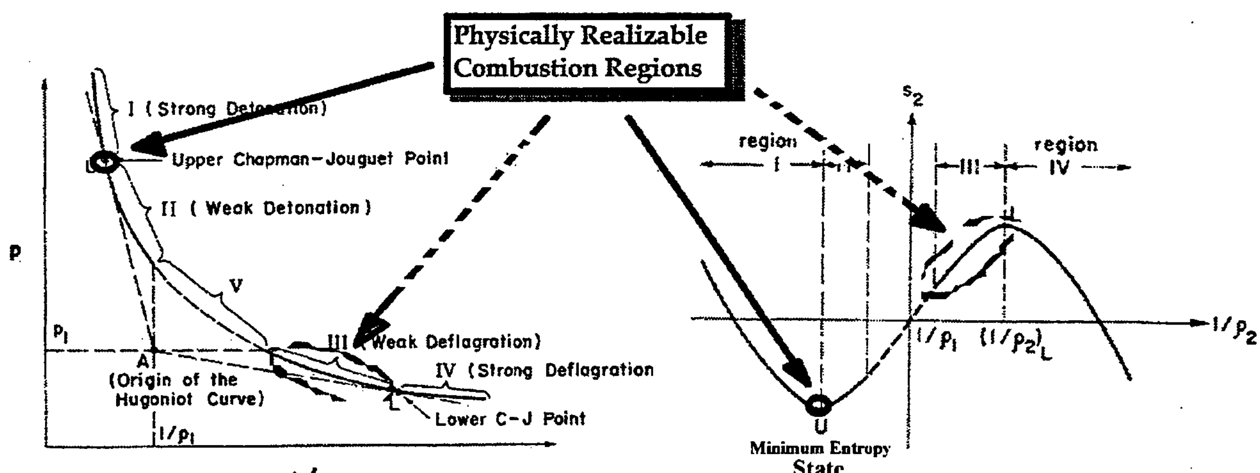

Figure 6. **Hugoniot Curve and Resulting Entropy for Each Point on the Curve** [4]

The C-J points, described earlier, are the boundaries for strong and weak combustion events. The C-J points are the two physical solutions to the Hugoniot relation for the constant area geometry shown in Fig. 5. The solutions are defined by the intersection of the Rayleigh line with the Hugoniot curve. The Rayleigh line condition is a thermodynamic limitation. For constant area combustion tubes, the maximum flow velocity is limited to the sonic velocity of the burned gas

(thermal choking). Although the detonation wave propagates and consumes the reactants at supersonic velocity, the flow of the combustion products "away" from the detonation wave is limited to Mach 1, relative to the wave. This is one of the major conditions that determine the location of the upper and lower C-J points. Since the Rayleigh line is a straight line with a negative slope passing through the origin of the Hugoniot curve (Point A), the acceptable end states are divided into two distinct branches: the upper detonation branch and the lower deflagration branch.

4.2 PDE PROPULSION

The objective of any propulsion system is to minimize the entropy rise in the working fluid. Figure 6 shows that for a given specified initial condition, a detonation results in the lowest possible entropy state on the Hugoniot curve. PDE and PDRE performance gains over conventional engine cycles can be realized if the entropy gain shown in Fig. 6 can be achieved in practical engine designs, and if transient operational issues are appropriately addressed.

Detonation combustion is an efficient means of burning propellant mixtures to release the chemical energy content. The very rapid energy conversion associated with detonation combustion can lead to more compact and efficient propulsion system designs relative to conventional systems presently in service. Researchers have recognized the performance potential of detonation combustion for over 75 years. However, pulse detonation propulsion technology has been slow to mature due to the difficulties involved with rapidly and reliably injecting, mixing, and igniting gaseous and liquid propellants, controlling transition to detonation, and exhausting combustion products on a time scale such that the entire process can be repeated in milliseconds [18, 19, 20].

Designs for intermittent flow propulsion devices were envisioned as early as the turn of the 20th century. German scientists developed many intermittent flow jet-propulsion engine designs beginning as early as 1900 but were never successful in developing a true constant-volume propulsion device. German designs included "explosion cycle" and "pulsejet" engines, both of which are significantly less efficient than a true constant-volume propulsion device. Performance of the explosion cycle and pulsejet engines is limited by the slow reaction rate, which limits operating frequency, pressure rise, and specific impulse. In many of the early designs the frequency of operation was dependent on the timing of the mixture and ignition arrangement. These engines did not necessarily operate at the natural frequency of the system, resulting in their classification as non-resonator-engines. Other designs were classified as resonator types wherein the operating frequency was tuned to the acoustic resonances of the combustion chamber [21]. The V-1 "Buzz-Bomb" which entered service in 1944 is an example of a resonator type pulsejet. Continued efforts led to development and test of valveless engine designs that operated at very high frequencies [21]. It is unclear from the literature why the technology failed to receive continued development attention, given the demonstrated performance levels and the simplistic, lightweight design of the propulsion devices.

The U.S. Navy, Office of Naval Research (ONR) initiated Project Squid shortly after World War II to investigate the performance of pulsejet engine designs for military and commercial systems [7]. The work was initiated with propulsion assets and information captured from the Germans at the conclusion of the war. These pulsejet engines were shown not to be detonation engines since their combustion processes occurred with subsonic flame speed in a resonant cavity. The U.S. effort was eventually terminated since it was determined that the propulsion system did not have a high overall efficiency [7].

By the mid-1980's U.S. researchers were able to demonstrate higher-performance with sustained detonations at moderate operating frequencies. These modern efforts have led to renewed interest in developing PDE and PDRE technology due to the potential for improvement in thermodynamic efficiency and performance, relative to existing systems, and the emergence of many potential military and commercial system applications.

Dr. Schmuel Eidelman introduced the recent revival of pulse detonation engine research by demonstrating an experimental pulsed combustion device in 1986 [22]. This work was conducted at the Naval Postgraduate School and was sponsored by the Office of Naval Research. During this study fundamentally new elements were introduced that distinguished the NPS concept from previous work. This effort was the first successful demonstration of a self-aspirating pulsed combustion device. In addition, the operating frequency was synchronized with that of the fuel mixture injection by timing the fuel valve opening and spark ignition, thus establishing the feasibility of intermittent injection. Initial work consisted of establishing single detonations in a chamber containing ethylene and air using an ethylene-oxygen pre-detonator. Additional work resulted in demonstration of repetitive detonations. Periodic fuel injection within the naturally aspirated chamber resulted in a maximum operating frequency of 25 Hz. The specific impulse estimated using the pressure-time history and the amount of fuel consumed ranged from 1000-1400 seconds [22]. Subsequent analysis of this work by Kailasanath suggests the velocities of the observed detonation waves were significantly below the C-J detonation velocities for the reported mixtures, indicating that a fully developed detonation wave was not formed [18]. However, the results of Eidelman, Helman, and Shreeve led to continued research with this concept and influenced modern PDE and PDRE development efforts [18].

A brief history of the development of detonation theory is provided in Reference 7. An overview of early detonation wave research is provided in Reference 13. A detailed overview of the status of experimental and theoretical research on pulse detonation engines is also provided in reference 18. Kailasanath reviews early attempts to use detonations for propulsion and discusses the possible reasons for success or failure of these experimental works. In addition, Kailasanath reviews recent experimental work, draws observations from the results, and discusses possible implications of these results on future PDE development efforts.

The Department of Defense (DoD) and NASA are presently sponsoring fundamental research and exploratory development programs that will establish a basis for demonstration of prototype PDE and PDRE systems. U.S. industry firms and academia are contributing to the government-sponsored programs, and the U.S. industry is also supporting development of the technology with corporate funds. Active pulse detonation propulsion programs are underway in France, Canada, Russia, Belgium, and Israel, and Japan, Norway, China, and Poland are performing PDE-related work. A detailed discussion of U.S. technology programs is provided in Section 10.0 of this report.

5.0 CHARACTERISTICS OF COMBUSTION PROPULSION CYCLES

The motivation for pursuing development of pulse detonation propulsion technology, as mentioned previously, resides in the inherent thermal efficiency advantage associated with the detonation cycle. The high thermodynamic efficiency of the detonation cycle is traceable to the low entropy rise in the working fluid. A brief comparison of deflagration and detonation combustion cycle efficiencies is provided in the following paragraphs.

Deflagration combustion in conventional airbreathing and rocket engines occurs under nearly constant pressure conditions. Deflagration reactions propagate at relatively low flame speeds. The flame speed is governed by the laminar or turbulent diffusion of unburned gases ahead of the flame and burned gases behind the flame. Typical wave speeds for a deflagration combustion wave range from 1-30 m/s [7]. Deflagrations produce small decreases in pressure and can be modeled as nearly isobaric, or constant pressure, processes [8, 23]. Propellant feed systems for these propulsion devices are required to deliver the fuel and oxidizers to the combustion chamber at elevated pressures in order to achieve desired thrust levels.

PDEs and PDREs rely on periodic, cyclical detonation of fuel/air and fuel/oxidizer mixtures to produce thrust. A detonation is a supersonic combustion wave that typically propagates at a few

thousand meters per second relative to an unburned fuel-air mixture. Detonation is a much more dynamic and violent phenomenon than deflagration and produces large overpressures. A detonation wave compresses the reactants, increasing their pressure, density, and temperature. Detonations can be modeled as supersonic shock waves that initiate and are closely coupled to a thin flame front of the combustion region. Due to the high-speed nature of a detonation wave, detonation closely approximates a constant volume combustion process [8, 23, 24].

Figure 7 provides a comparison of the pressure-specific volume (Fig. 7A), and temperature-entropy (Fig. 7B) characteristics of the Brayton and Humphrey cycles. The ideal Brayton and Humphrey cycles are similar in that both use isentropic compression and expansion processes to transfer work to and from the system. The Brayton cycle represents the constant pressure heat addition of deflagration combustion. The Humphrey cycle represents the constant volume heat addition of the detonation combustion process. The Brayton cycle (0-1-4-5-0) consists of two constant pressure processes (1-4 and 5-0) and two isentropic processes (0-1 and 4-5). The Humphrey cycle is similar, except that the constant pressure combustion process of the Brayton cycle, (1-4), is replaced by a constant volume heat addition process (1-2). The total area under the Humphrey P-v curve is greater than the total area under the Brayton P-v curve, indicating a greater availability of useful work from the Humphrey cycle [8, 23, 24].

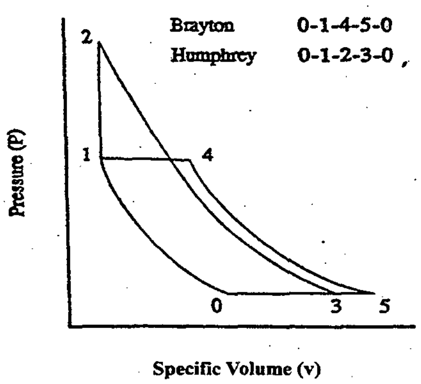

Figure 7A. Pressure-Specific Volume Cycle Diagram [8]

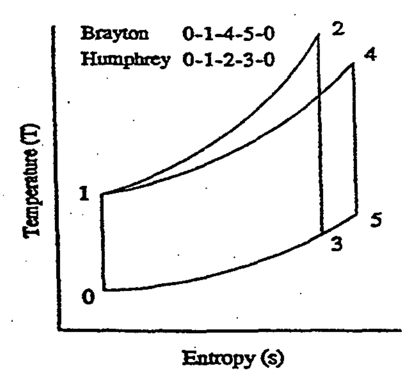

Figure 7B. Temperature-Entropy Cycle Diagram [8]

The efficiencies of the constant pressure Brayton cycle and the constant volume Humphrey cycle can be computed from the pressure-volume and temperature-entropy diagrams shown in Figs. 7A and 7B. The efficiency of a cycle is defined as the useful work output divided by the total heat energy input [24].

The efficiency of the Brayton cycle depends only on the temperature change during either of the two isentropic compression or expansion processes (i.e., To/T1 = T4/T5):

$$\eta_{BRAYTON} = 1 - \frac{T_0}{T_1} \quad (8)$$

The efficiency of the Humphrey cycle is given as:

$$\eta_{HUMPHREY} = 1 - \gamma \frac{T_0}{T_1} \left[\frac{\left(\frac{T_2}{T_1} \right)^{\frac{1}{\gamma}} - 1}{\frac{T_2}{T_1} - 1} \right] \tag{9}$$

The efficiency of the Humphrey cycle depends not only on the isentropic compression temperature ratio, To/T1, but also on the ratio of specific heats, γ, and the temperature change due to the constant volume combustion (i.e., the detonation temperature ratio T2/T1).

The difference between the Brayton and Humphrey cycle efficiencies is the following To/T1 multiplier:

$$\gamma \left[\frac{\left(\frac{T_2}{T_1} \right)^{\frac{1}{\gamma}} - 1}{\frac{T_2}{T_1} - 1} \right] \tag{10}$$

The value of this expression is always less than one for detonation combustion. As a result, the efficiency of a Humphrey (detonation) cycle is greater than the efficiency of the Brayton (deflagration) cycle. For additional thermodynamic cycle analysis, see references 8, 23, and 24. Reference 8 also provides a detailed description of detonation physics and detonation wave modeling.

A comparison of Brayton and Humphrey cycle efficiencies can be made using a representative detonation combustion process and approximating the ratio of specific heats (γ) as a constant throughout the cycle. An equilibrium chemistry calculation for stoichiometric hydrogen/air at atmospheric conditions yields a detonation temperature ratio T_2/T_1 of 10.2 and specific heat ratios of 1.4 in the unburned gas and 1.16 in the burned gas. The Humphrey detonation cycle efficiency can be calculated for average cycle γ's of 1.4 and 1.16, which represent the upper and lower limits of the varying specific heat ratio.

Figure 8 provides the calculation of cycle thermal efficiency as a function of compression ratio, P_1/P_0. The actual detonation cycle efficiency lies somewhere between the two limiting specific heat curves ($\gamma = 1.4$ and $\gamma = 1.16$). At a compression ratio of 6, for example, the constant volume process offers a 30 to 50% improvement in thermal cycle efficiency over the constant pressure cycle[8, 23].

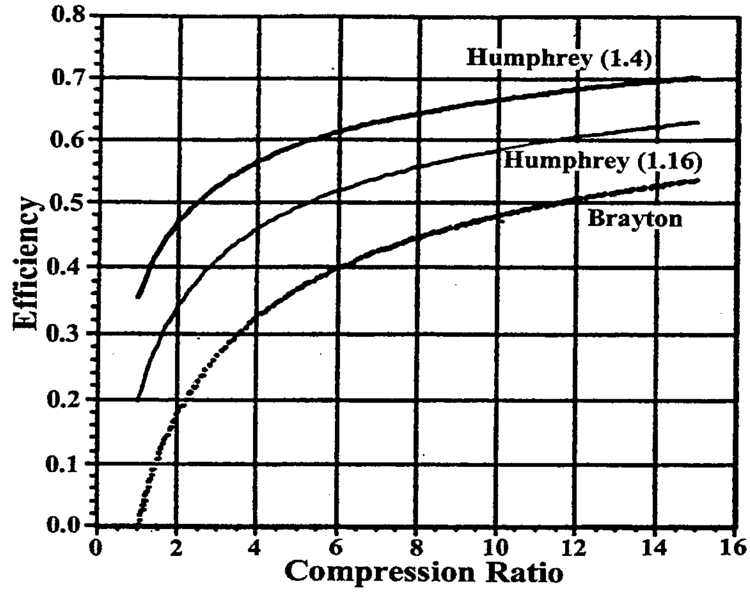

Figure 8. Thermodynamic Efficiency of Brayton Isobaric and Humphrey Detonation Cycles for Stoichiometric Hydrogen/Air [8]

A constant pressure engine cycle is compared to a constant volume and a true C-J detonation cycle in Fig. 9. For purposes of comparison, the only process that is different in the three cycles is the method of energy conversion or heat addition.

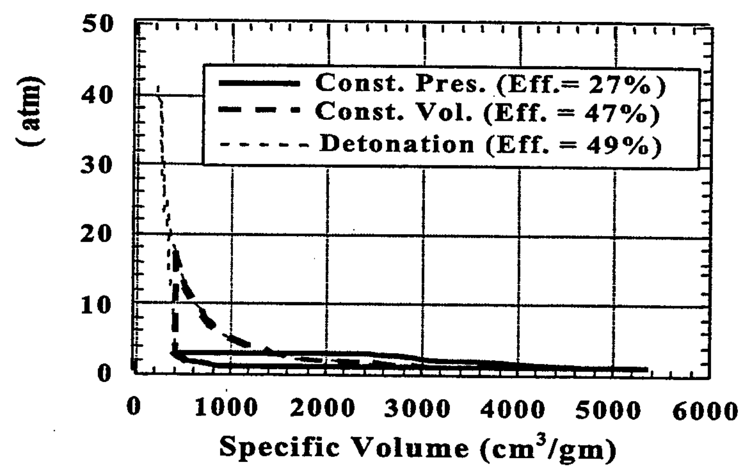

Figure 9. Thermodynamic Efficiency of Brayton, Humphrey, and Chapman-Jouguet Detonation Cycles For Stoichiometric Hydrocarbon/Air [25]

The amount of heat added is also kept the same at 50 kcal/mole (a value typical of hydrocarbon fuels) for the three cycles. In all cases, the fuel-air mixture initially compresses adiabatically from 1

to 3 atm. before heat addition. After heat addition, the products of combustion are expanded adiabatically to 1 atm. Finally, the system is returned to its initial state. Since all processes except heat addition are the same, the relative thermodynamic efficiency of the three combustion processes can be obtained by comparing the areas enclosed by the curves. The thermodynamic efficiencies for the three cycles are: 27% for constant pressure, 47% for constant volume, and 49% for detonation [25]. Thus, the thermodynamic efficiency of the detonation cycle is close to that of the constant volume cycle, and significantly better than that of the constant pressure cycle. A major challenge in the development of pulse detonation propulsion systems is attaining this higher potential efficiency in practical propulsion devices. A PDE or PDRE must possess a high propulsion efficiency to benefit from the high thermodynamic efficiency of the constant volume combustion cycle [25].

Although the constant volume cycle shows a significant efficiency advantage in both cases, this comparison cannot be taken as the correct quantitative system comparison because pulse detonation propulsion devices operate in a pulsed transient mode. However, this comparison does indicate that one begins with a much more efficient cycle to develop pulse detonation propulsion technology [19].

6.0 PDE/PDRE CYCLE OPERATION

Figure 10 shows pictorially the events occurring in a single detonation cycle for a tube with one end closed and one end open. The detonation is initiated near the closed end. The cycle begins with the empty chamber shown in Fig. 10A. Continuing clockwise, a combustible fuel-air or fuel-oxidizer mixture is injected at the valved end (closed end) of the tube at pressure P_1 and temperature T_1 in Fig. 10B. P_1 and T_1 are determined by flight conditions and inlet design characteristics for airbreathing engines, and the chamber inlet pressure may be as low as 200 psi for PDRE systems.

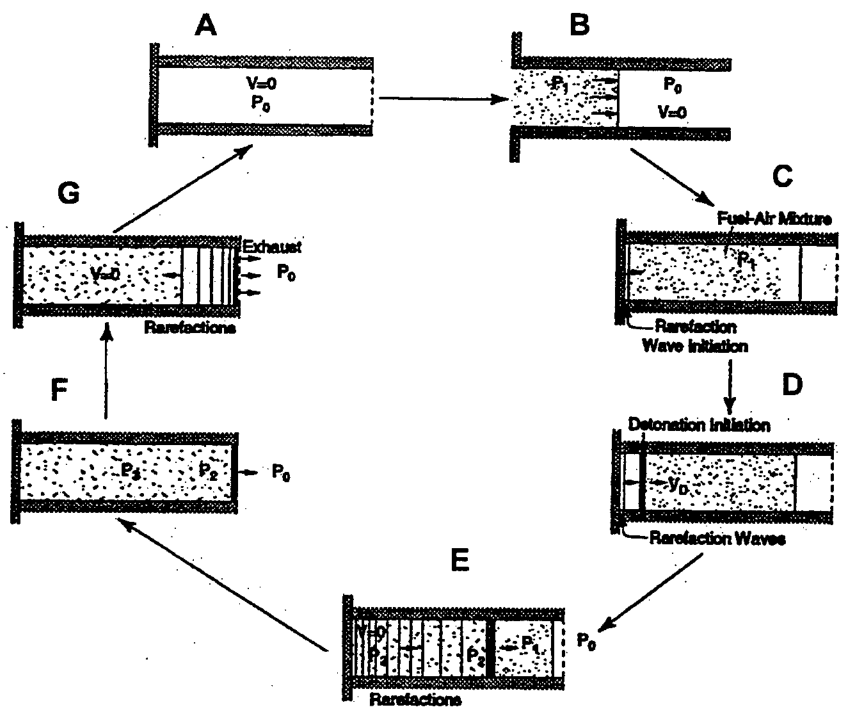

Figure 10. Pulsed Detonation Cycle Operation [8]

Continuing clockwise, the fuel and oxidizer valves are closed (Fig. 10C) once a full propellant load has been injected, and the combustor exit remains open. Valve timing ensures that the propellant mixture and the detonation wave reach the combustor exit simultaneously to prevent any unburned gases from escaping and lowering the operational efficiency.

Once the fuel and oxidizer valves are closed, an ignition source initiates a deflagration near the closed end of the chamber that quickly transitions to detonation (Fig. 10D). An expansion zone is created between the closed end of the chamber and the detonation wave since the velocity must be zero at the closed end. Rarefaction waves are generated at the closed end of the chamber and proceed towards open end of the chamber. The rarefactions originate at the closed end and satisfy the constraint of zero axial fluid velocity normal to the wall. The strength of the expansion region is a function of γ, the ratio of specific heats of the burnt gases. The ratio of specific heats of the combustion products determines the axial velocity of the burned gases behind the detonation wave, which must be decelerated to satisfy the closed end boundary condition.

Once the detonation is initiated the detonation wave propagates towards the open end of the chamber (Fig. 10E). Ideally, the detonation wave will proceed at the C-J detonation velocity of the mixture, V_D. The region in front of the detonation will contain the unburned mixture at state 1. The burned mixture immediately behind the detonation wave will be at significantly higher temperature and pressure, state 2, as discussed previously in Section 2. The burned mixture near the closed end of the chamber will be at state 3, much lower in temperature and pressure than state 2 due to the expansion region (see Fig. 2).

As the detonation wave exits the chamber, the chamber will contain combustion products at elevated temperatures and pressures (Fig. 10F). Conditions along the length of the chamber range from T_3 and P_3 at the closed end to P_2 and T_2 at the open end. The axial velocity of the combustion products varies from zero at the closed end to supersonic values outside the chamber exit.

As the detonation wave exits the chamber, a pressure differential exists at the open end. This pressure difference creates a series of rarefaction waves that propagate back into the chamber helping to expel the combustion products. The rarefaction waves travel into the chamber at the speed of sound of the combustion products.

After the primary combustion products are expelled from the chamber, the remaining gas within the chamber is at a pressure near P_3 (Fig. 10G). The unsteady blowdown (expansion) process is characterized by a series of compression and rarefaction waves that are alternately created and reflected and accelerate the burned gas towards the open end of the chamber. The blowdown process is self-aspirating, and the flowfield at the combustor exit alternates between outflow and inflow. The pressure and temperature eventually decay to ambient levels and the exhaust velocity decays to zero [8].

Once the pressure within the chamber drops to appropriate levels, the chamber is recharged with a fresh fuel-air or fuel-oxidizer chamber load and detonation is initiated once again. The cycle frequency for a pulsed detonation propulsion system is the inverse of the time required to complete a full detonation cycle: $T_{CYCLE} = T_{DETONATION} + T_{BLOWDOWN\ (EXPANSION)} + T_{FILL}$ [8]. PDEs and PDRE may also require active purging of the residual combustion products from the chamber prior to refilling the chamber in order to avoid premature ignition of the fresh propellant charge.

As mentioned previously, many experimental efforts are presently underway to characterize optimum combustor geometry, ignition location, fuel detonation properties, ignition delay, and deflagration-to-detonation transition properties using single-chamber test apparatus with cycle times

ranging from low to moderate frequencies [26, 27, 28]. Other technology programs are proceeding with development and testing of PDE and multi-chamber PDRE components and subsystems [29, 30, 31]. Engines with multiple combustion chambers will make use of fast acting propellant metering valves to sequentially load the chambers with propellant to increase aggregate operating frequencies. The objectives of component and subsystem development programs include demonstration of operation, performance, and throttling of flight-scale components. Additional discussion of current technology development efforts is provided in Section 10 of this report. Technological challenges associated with implementation of the pulse detonation cycle in an operational system are briefly discussed in Section 11.

7.0 CONCEPTUAL PDE/PDRE SYSTEM DESIGNS

7.1 PULSE DETONATION ENGINES (PDEs)

Pulse detonation engine technology is still in an early stage of development. Several technical challenges must be successfully addressed before operable engines become a reality. PDEs cyclically detonate onboard fuel and atmospheric air mixtures to generate thrust. Major PDE subsystems include inlets, detonation chambers, and nozzles. Practical PDEs may include several detonation chambers fed by a common inlet and exhausting through a common nozzle flowpath. In addition to these major subsystems, PDEs require pressurized fuel storage and feed systems, fuel/air injection systems, and detonation initiation systems. PDE propellant injection systems will include high-speed fuel and air metering valves to cyclically load detonation chambers with fresh propellant charges at the beginning of each combustion cycle. Detonation pressure forces acting on the closed (valved) end of the PDE detonation chambers (thrust walls) convert chemical energy into kinetic energy. PDEs will require auxiliary power systems for detonation initiation and flow control, and may include power extraction systems for certain applications.

Rapid and reliable initiation of detonation is one of the major challenges that must be addressed before operational PDEs become a reality. The ability to rapidly and reliably initiate detonations with practical fuels and initiation energy levels is critical to successful development of PDEs, as very high operating frequencies and repeatable ignition times are fundamental engine operating requirements. Initiator units, or pre-detonators, may be employed to ensure reliable, repeatable detonation and also minimize engine weight, packaging volume, and detonation cycle time. In addition, reliable fuel/air mixing techniques are required to ensure propellant mixtures in the main detonation chambers are within the detonability limits of the selected propellant combination.

Initiator units make use of onboard fuel, high-density oxidizer, and air to establish deflagrations that rapidly transition to detonation. Initiator unit components include onboard oxidizer storage tank, oxidizer, fuel, and air feed system components, and small diameter detonation initiation chambers. Operationally, initiator units work as follows: a deflagration is initiated in a small-diameter detonation chamber filled with an oxidizer/fuel/air mixture, the high-speed deflagration rapidly transitions to detonation, minimizing transition length, and finally, the detonation combustion wave transitions from the small diameter chamber into the main detonation chamber filled with the less-detonable fuel/air mixture. Once detonation is established in the main detonation chamber the detonation transitions the length of the chamber adiabatically compressing and combusting the reactants. The high-pressure ratio of the detonation combustion reaction imparts thrust to the thrust wall, thereby transferring chemical energy to kinetic energy.

Development of direct initiation methods may reduce system weight, cost, and complexity even further through elimination of high-density oxidizer storage, feed, and mixing systems. However, direct initiation of the fuel/air mixture must be accomplished with practical energy-input levels, which may require development of new high-energy density fuels with exceptional detonability characteristics when the fuel-spray droplets are mixed with air. Direct initiation of detonation has been discussed for PDE applications. However, extremely high energy levels (thousands of joules)

are required for direct ignition of liquid fuel-air mixtures. Therefore, initiator units employing the DDT process are the leading technology option for near-term engine developments. Fuel/oxygen initiators have very short DDT lengths, short ignition delays, and are very reliable, and are therefore very attractive for initiation applications.

Additionally, design of PDE inlets and nozzles that operate efficiently over a range of flight conditions are required for successful, practical application of PDEs. PDE designs and configurations may vary widely depending upon design solutions adopted to overcome these technical challenges.

Fuel selection is driven by specific mission applications. High-speed aircraft applications may favor hydrogen fuel, whereas volume and weight limitations of missile systems may favor higher density liquid fuels. Performance optimization drives PDE designs to very high operating frequencies because thrust scales with frequency [34]. It has been estimated that in order for the PDE cycle to be competitive with conventional turbojet/turboramjet systems, they will be required to operate in the 75 to 100 Hz range with near stoichiometric fuel/air mixtures [35]. This represents a cycle time of approximately 10 msec. A 10 msec. detonation cycle requires fast acting propellant valves to fill/refill detonation chambers with fresh propellant charges within a 5 msec. time span, allowing approximately 2 msec. for detonation wave formation and propagation along the length of the detonation chamber, and approximately 3 msec. for expansion of rarefaction waves and chamber purging[35].

A schematic of a conceptual, single detonation chamber PDE is provided in Fig. 11. A brief description of the various components and their operation is provided in the following paragraphs.

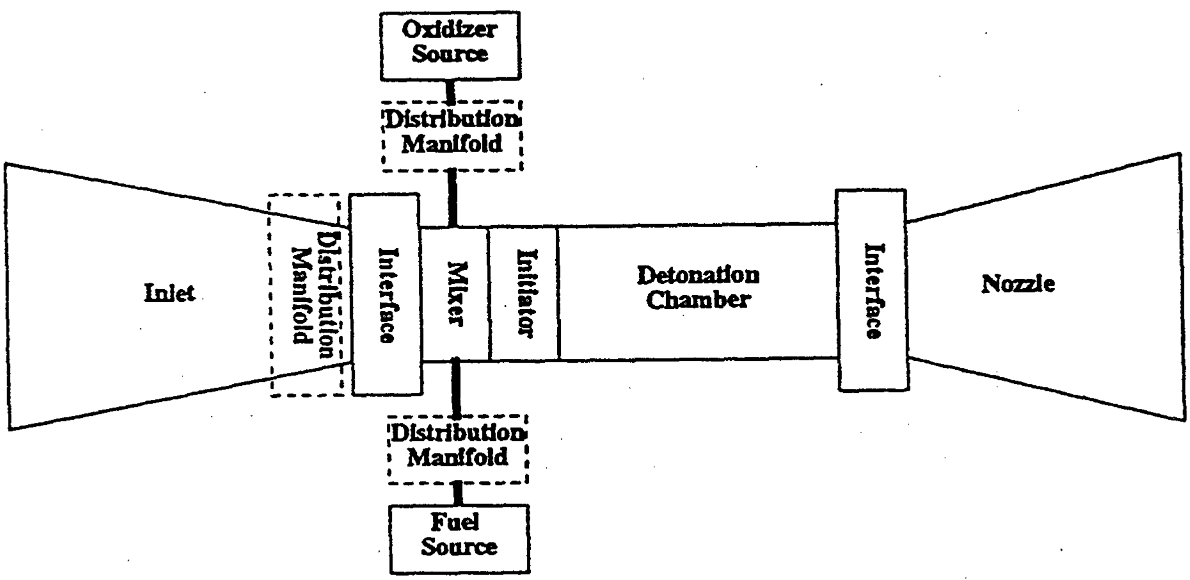

Figure 11. Conceptual Pulse Detonation Engine (PDE) Schematic [32]

The conceptual PDE system shown includes: 1) air inlet; 2) fuel source; 3) oxidizer source; 4) fuel/air/oxidizer distribution manifolds; 5) fuel/air/oxidizer mixing section; 6) inlet/detonation chamber interface; 7) initiator unit; 8) detonation chamber; 9) detonation chamber/nozzle interface; and 10) nozzle. In addition, engine power and control system components will also be required. Power extraction components may be required for some engine applications. However, due to the early stage of PDE technology development, power extraction techniques are not well formulated.

Design and development of PDE inlets that operate over a range of flight conditions introduce non-trivial engineering challenges. PDE inlets need to be isolated in some manner from the non-steady combustion process in order to maintain high operating efficiencies. Unsteady inlets, where some of the combustion products from the combustion cycle are expelled through the open inlet, have seen limited use, most notably with V-1 pulse jet engines[32]. More recently, researchers have evaluated unsteady inlets with pulse detonation devices[36-38]. Several unsteady, valveless, steady, and mixed compression inlet designs have been proposed and analyzed, [35-40] but detailed design, development, and integration of efficient PDE inlets remains a critical enabling technology.

Inlet flow must be separated from the cyclical detonation processes. Bussing briefly discusses aerodynamic and mechanical methods that have been employed or proposed to isolate the detonation process from inlet flow in Reference 32. In addition, Bussing introduces a rotary valve solution that can be used with a multiple combustor PDE concept in Reference 41. The rotary valve serves to isolate the steady operation of the inlet and fuel supply systems from the unsteady combustion processes. The rotary valve concept cyclically recharges some of the detonation chambers with fuel and air while detonations occur in other chambers, allowing the inlet and fuel systems to operate in a steady state mode. Other methods proposed to reduce or eliminate flow perturbations in the inlet flow include incorporation of high bypass ratio inlets, oversized inlets with fast-acting bypass valves, and pressure-activated flapper valves.

A key issue in the pulse detonation engine concept is the design of the main detonation chamber. The detonation chamber geometry contributes to the overall propulsion efficiency of the engine and determines the duration of the detonation cycle. The detonation chamber must be structurally designed to withstand the pressures and temperatures of the cyclic detonation process, and the design must also include a mechanism for transferring thrust loads to the vehicle. Refueling and purging strategies may also influence the design of the detonation chamber. Because the oxidant (air) is obtained from the external flowfield, the overall propulsion efficiency of the engine is dependent upon the interaction of the surrounding flow with the internal flow dynamics of the inlet and detonation chamber[32, 33, 36].

The PDE initiator unit creates self-sustaining detonations that travel into the main detonation chamber to initiate detonations in primary fuel/air mixtures. As discussed above, the initiator unit combines onboard fuel and oxidizer with atmospheric air to establish deflagrations that rapidly transition to detonation [32]. Multi-chamber PDEs may require multiple initiator units to maintain high aggregate detonation cycle frequencies. Initiator units, or pre-detonators, have been investigated in a number of studies[22, 42, 43].

PDE nozzle design and integration introduces challenges not present in conventional steady state combustion engines. PDE nozzle design and development remains a critical enabling technology. The purpose of a nozzle is to extract kinetic energy from thermal energy by expanding the flow of hot combustion products through a choked point, thereby causing the flow of hot gas to achieve supersonic flow velocities. The nozzle maintains backpressure within the detonation chamber, thereby ensuring detonation of propellant mixtures at elevated pressures to maintain engine operating efficiency. The challenge for pulsed combustion nozzle designers is to maintain choked flow at the nozzle throat with unsteady combustion processes occurring upstream of the nozzle. Unsteady nozzle inlet pressure requires that the nozzle operate over a large ΔP range. Unsteady nozzle flow subjects nozzles to unsteady thermal and mechanical loads, which complicate design. Delaval (converging-diverging) nozzles may result in unwanted reflection of detonation waves back into the detonation chamber. Reflected shocks can perturb combustor flow and impart high mechanical loading on the nozzle itself. Nozzleless detonation chambers, chambers with simple diverging nozzles, or aerospike nozzles might eliminate this concern and may provide higher overall performance.

One of the attractive potential benefits of PDEs is the lack of a compression cycle. The high pressure ratios attainable with detonation combustion eliminate the need for fuels and oxidizers to be delivered to the detonation chamber(s) at very high pressures. Consequently, simple pressure-fed systems can be employed, requiring storage of onboard fuel at moderate pressures only. Similarly, onboard oxidant used in conjunction with initiator units may be stored in required quantities at moderate pressure levels.

Fuel/air/ oxidizer distribution manifolds deliver the various propellants to the detonation chamber at slightly elevated pressure. High-speed propellant valves meter proper proportions of these propellants into the detonation chamber to produce uniform detonable mixtures. Dedicated mixing schemes may be required to ensure uniform mixing of propellants so that each initiator unit and main detonation chamber propellant charge is reliably within the detonability limits of the specific mixture. The positive-pressure distribution manifolds work in conjunction with high-speed propellant injection valves and engine/ignitor control system components.

Fuel selection is critical for PDE operation because ignition characteristics have a dramatic impact on the detonation process. Fuel can be stored in several different forms but must be delivered to the fuel manifold as a gas, liquid, and/or solid of sufficiently small droplet/particle size to permit the formation of stable detonation. Solid particulate or liquid droplets must be very small to ensure that the droplet/particle combustion time is compatible with the detonation time scale[32].

Each of the PDE subsystems mentioned must meet certain cost, weight, and volume requirements in order to be of practical use. In addition, active subsystems must be able to operate with reasonable power requirements. Throttling capability may be necessary for some PDE applications, such as aircraft propulsion, planetary excursion vehicles, etc. PDE throttling may be achieved by varying propellant valve actuation rates and initiator frequency, or by other means depending upon the engine configuration.

7.2 PULSE DETONATION ROCKET ENGINES (PDREs)

Similar to PDEs, PDREs produce thrust by cyclically detonating propellants within a detonation chamber and exhausting the combustion products through a nozzle. However, unlike PDEs, PDREs must carry all of the oxidizer necessary to complete their specific missions onboard, and PDREs may have vacuum start and restart requirements for many applications. PDREs will incorporate propellant storage, feed, and injection system components, one or more detonation chambers, ignition systems, detonation chamber/nozzle interface hardware, nozzles, and engine control system components. Similar to PDEs, numerous PDRE configurations may evolve depending upon engineering design solutions adopted to overcome technical challenges.

Once propellants are injected into PDRE detonation chambers, the fast-acting propellant injection valves close to seal the detonation chamber and detonation is initiated. The detonation wave passes through the detonation chamber at supersonic velocities, igniting the propellants and elevating the upstream pressure to several times (6-12 times) that of the initial fill pressure[30]. Detonation wave residence time within the detonation chamber is on the order of 1-3 msec., depending on the thermodynamic conditions of the propellants and the detonation chamber geometry[30]. Once the detonation wave exits the chamber, a series of rarefaction waves propagate from the open end of the chamber towards the closed (valved) end, helping to expel residual combustion products and reduce the pressure within the chamber. Once the chamber pressure drops to a specified level, the chamber purge and refill operations can be initiated.

Like conventional, steady combustion rocket engines, PRDEs require pressurization of the detonation (combustion) chamber(s) to obtain high performance. However, due to the high pressure ratios associated with detonation combustion, PDRE detonation chamber fill pressures are

much lower than chamber pressures associated with conventional rocket engines[30]. PDREs can employ a variety of thermodynamic cycles to power turbopumps, such as gas generator, staged combustion, or expander cycles, or incorporate pressurized feed systems[30]. PDRE operating pressures may be on the order of 100-200 psia, as compared to several hundred psia for conventional open cycle engines and several thousand psia for conventional closed cycle engines.

The NASA Marshall Space Flight Center is currently sponsoring two PDRE research and development efforts. MSFC has awarded contracts to Pratt & Whitney Aerosciences and the United Technologies Research Center (UTRC) to develop and demonstrate alternative system designs. Pratt & Whitney Aerosciences has developed a concept where multiple detonation chambers exhaust through a common nozzle flowpath[30]. This chamber/nozzle arrangement provides the necessary backpressure to maintain desired chamber fill pressures and obtain adequate performance levels. UTRC is developing a single detonation chamber PDRE that incorporates a nozzle with an aerodynamic throat[40]. The variable boundary layer created by the aerodynamic throat ensures maintenance of adequate back pressure within the detonation chamber and allows the nozzle to operate in a quasi-steady mode, thereby ensuring adequate overall engine performance. Both of the MSFC-sponsored PDRE development engines require well-contoured nozzles to ensure that the nozzle flows remain attached to the nozzle walls.

The multiple-chamber PDRE concept shown in Fig. 12 is provided by Pratt & Whitney Aerosciences (formerly Adroit Systems, Inc.) [23]. This concept includes six detonation chambers coupled to a common feed system and nozzle assembly. Operationally, the detonation chambers are fired in a phased manner allowing the feed system and manifolds to operate in a steady state manner. All of the detonation chamber combustion products are exhausted through the single common nozzle.

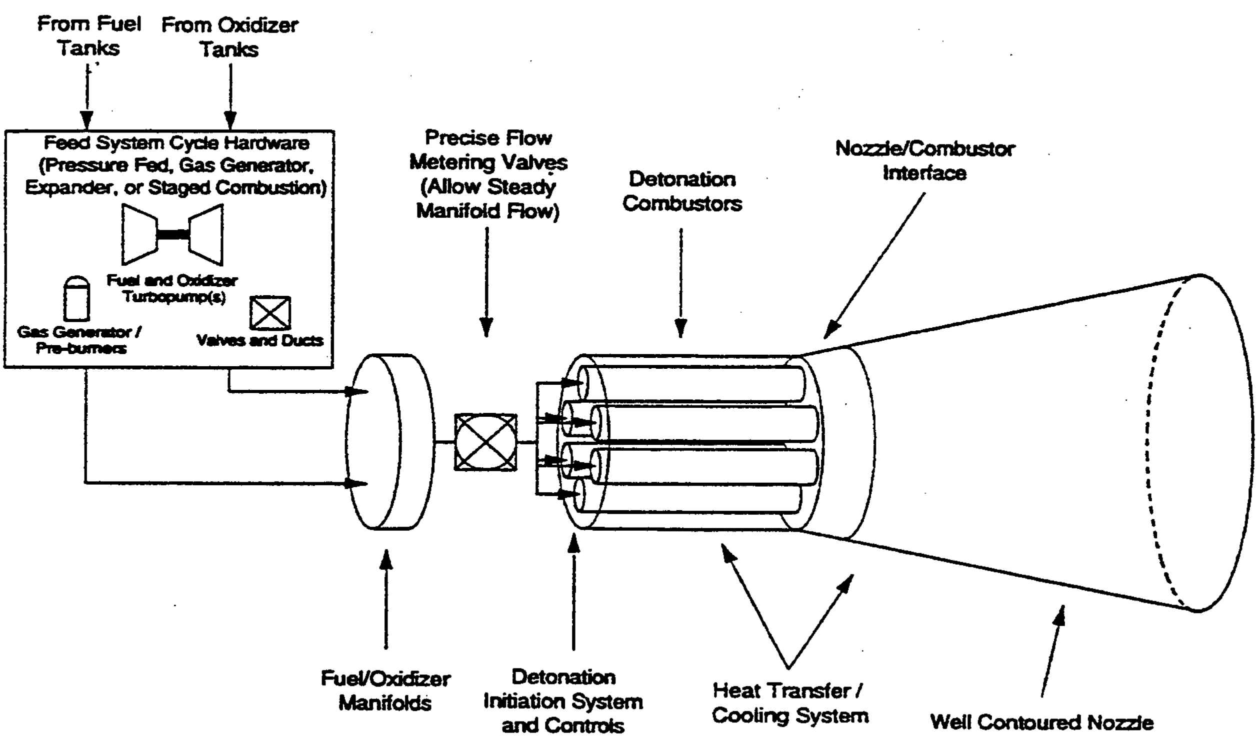

Figure 12. Conceptual Pulse Detonation Rocket Engine (PDRE) Schematic [44]

In addition to the major PDRE components and subsystems discussed, PDREs may also require gimbal mounts, thrust vector control actuators, thermal protection systems, and power conditioning systems[24]. PDRE designs need to incorporate methods to maintain choked flow at the nozzle throat

to maintain engine-operating efficiency, and the nozzle designs need to accommodate the cyclic thermal and mechanical loads to meet service life requirements. Multiple chamber PDREs exhausting into a single common nozzle face the additional design challenge of mitigating tube-to-tube interference to achieve operable design load environments. Current technology efforts suggest nozzle design and operation requires additional development effort.

Other PDRE configurations are possible. PDRE designs and configurations are influenced by specific mission requirements and engineering solutions adopted to address fundamental technical and operability issues.

8.0 SYSTEM MODELING

Although significant progress has been made in the U.S. regarding pulse detonation science and technology development, system and performance modeling requires additional effort. Various computational studies indicate a wide variation in predicted PDE performance with estimates ranging from 1100 seconds to 8000 seconds of specific impulse for stoichiometric hydrogen-air[25]. Pulse detonation system modeling offers new analytical and computational challenges not present in constant pressure combustion modeling, and very little experimental data on system performance has been reported in the open literature. Therefore, there is significant uncertainty in the actual performance of even an idealized laboratory-scale PDE. PDE/PDRE results from computational studies are strongly dependent on the fidelity of the physical model on which the equations are based, numerical resolution, initial conditions assumed for detonation initiation, specific geometry simulated, and boundary conditions[25, 45].

Numerous models of varying complexity have been employed. While none of the models have attempted to represent the unsteady, multi-phase, reactive, 3D flows in an engine, many capture some aspect of the essential physics in a 1D or 2D geometry[25].

One of the primary goals of current pulse detonation technology initiatives is the advancement and validation of system modeling tools. Many current models do not fully account for valve losses, mixing losses, and other "real engine" effects. Calculation of theoretical cycle efficiency requires prediction of detonation wave structure, and the resulting head-end pressure-time history, which is dependent upon combustor geometry. The geometry influences the evacuation and refilling times as well as the pressure history while the detonation wave traverses the chamber. Consideration of losses and combustor geometry may result in considerable differences in the calculation of pulse detonation cycle efficiencies. Varying assumptions and boundary conditions used in the problem formulation result in varying performance predictions.

An example of performance model variation is provided in Fig. 13, which shows the variation in PDE specific impulse for instantaneous versus gradual relaxation of combustion gases once the detonation wave exits a chamber of the same geometry[19].

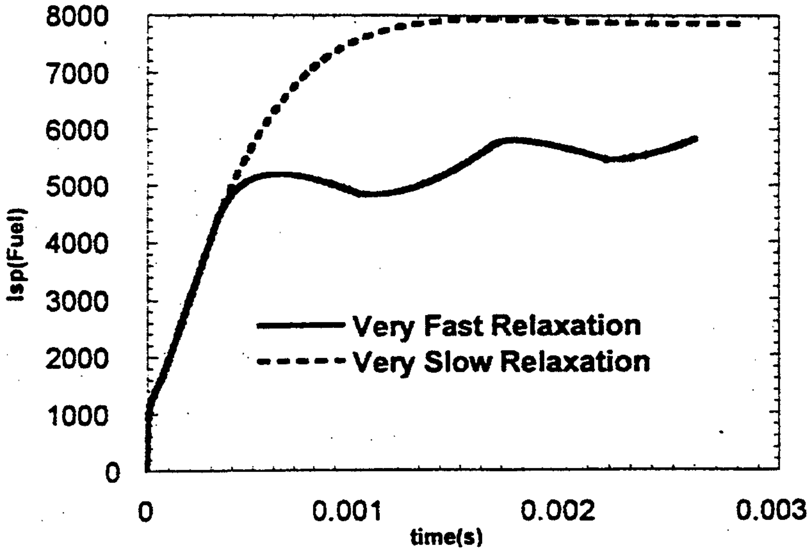

Figure 13. Variation of Isp With Combustor Relaxation Mode [19]

9.0 PULSE DETONATION PROPULSION SYSTEM APPLICATIONS

Flight vehicle main propulsion systems can usually be categorized into one of the following general categories: airbreathing, rocket, combined-cycle, dual-mode, or hybrid. Many potential applications exist for airbreathing and rocket pulse detonation propulsion devices. If successfully developed, PDEs could be used to power tactical aircraft, air-launched and ship-launched missiles, unmanned aerial vehicles, and a wide range of stand-off munitions. PDREs could be used to power space launch vehicle upper stages, orbit transfer vehicles, excursion vehicles and planetary landers. PDREs may also be used for spacecraft attitude control, satellite station keeping, and satellite maneuvering propulsion.

Combined-cycle systems place components of differing engine cycles in the same flowpath. A PDE placed in the same flowpath with a conventional rocket is an example of a combined-cycle system that might be used to power a high-speed, long-range missile. This type of system would require less packaging volume than a conventional two-stage liquid fueled rocket intended for the same mission. A PDE/ramjet/scramjet combined-cycle system might be used to power a hypersonic flight vehicle. The PDE would be used as the low-speed accelerator and hand over powered flight to the ramjet after achieving a flight velocity in excess of Mach 3.

A variation of the combined-cycle system is to share hardware for two differing engine cycles in the same flowpath. This variation of the combined cycle is termed dual-mode. Dual-mode applications for PDE and PDRE systems are not currently well defined.

There are many potential applications for PDEs in hybrid systems. Hybrids are defined here as any combination of a PDE with turbomachinery. In the hybrid mode, a PDE can be used in place of high-pressure compressor stages, combustion chambers, high-pressure turbine stages, and afterburners (or augmentors). For a given air flow, a PDE would provide an approximate 2-fold increase in overall pressure ratio on a time-averaged basis due to the detonation wave compression process. PDEs used for thrust augmentation will likely improve performance and reduce fuel requirements relative to current augmentor configurations.

Pulse detonation engines can also be considered for propulsion systems using combination cycles, i.e. when two or more engine cycles are used but do not interact. An example of a PDE combination cycle is use of a PDE as a duct burner. Likewise, for access to space applications, PDEs can be mounted in engine bays separate from the vehicle underside scramjet flowpath.

10.0 U.S. PULSE DETONATION TECHNOLOGY INITIATIVES

10.1 OVERVIEW

Recent advances in high-frequency pulse detonation propulsion technology by a number of different organizations have resulted in renewed interest in development of the technology for a broad range of propulsion applications[18-20, 22,-24, 26-49]. The DoD and NASA are currently sponsoring a number of fundamental research and exploratory development activities to advance the state-of-the-art[19, 44-49]. Academia and industry are contributing heavily to each of the government initiatives with the objective propulsion technologies targeted towards high-speed missile, aircraft, access to space, and space applications. NASA's primary technology application areas include subsonic and supersonic commercial aviation, access to space, and space exploration missions[49]. If successfully developed, the DoD could apply PDE technology to a broad range of application areas, including tactical aircraft propulsion, missile propulsion, space launch vehicle upper stage propulsion, advanced space vehicle propulsion, and spacecraft attitude control, maneuvering, and station keeping propulsion.

Recent PDE technology advances have led to the formation of two U.S. industry teams that are participating in many of the government-sponsored initiatives. The industry teams are also pursuing independent technology development activities due to the high potential payoff for commercial, military, and aerospace applications of pulse detonation technology should the technology be successfully developed. The industry teams have formed to pull together the range of propulsion, flight dynamics, and systems expertise to move the technology forward quickly. At present, the industry teaming arrangements are Pratt & Whitney Aerosciences/The Boeing Company/Pratt & Whitney/United Technologies Research Center (UTRC), and General Electric/Science Applications International Corporation (SAIC)/Advanced Projects Research Incorporated (APRI). Commercial industry objectives include resolution of existing technical challenges and development and demonstration of flightweight airbreathing, rocket, and hybrid pulse detonation propulsion systems through corporate and government-sponsored technology programs.

DoD and NASA are presently sponsoring theoretical and experimental research relating to all aspects of pulse detonation science and technology development. Current activities support PDE, PDRE, combined-cycle, and hybrid propulsion system development objectives. Currently funded U.S. Government initiatives and their objectives are discussed below.

10.2 DEPARTMENT OF DEFENSE

10.21 NAVY

Successful development of PDE technology could result in its adaptation to many Naval weapons systems. Navy shipboard and aircraft missile systems are necessarily volume and weight limited. Currently, most tactical missiles employ solid rocket motors due to their simplicity and high-speed capability, but they have limited range. Turbojets and turbofans are used for missiles requiring longer range or heavier payloads due to their high specific impulse, but these systems become expensive for high Mach number missions ($M=2\text{-}3$). For long range at higher speeds ($M=2\text{-}4$), ramjets and ducted rockets have been developed. However, they require solid rocket boosters to accelerate them to ramjet take over speed, which increases their cost, complexity, and propulsion system volume. Combined cycle engines, such as turbo-rockets and turbo-ramjets have also been

considered for missions requiring wide ranges in operating speed, but they are also more complex and more expensive[19, 26].

Application of the pulsed detonation cycle potentially offers Navy systems increased range, stealth, and reliability for systems in the Mach 0-3 operating range, while simultaneously offering reductions in size, vulnerability, and cost. In addition, multi-chamber PDEs may enable fluidic thrust vectoring, thereby eliminating the need for heavy, high-drag aerodynamic control surfaces. The Navy may want to continue the use of hydrocarbon fuels, such as JP-10, hydrocarbon blends, or high density strained hydrocarbon fuels to satisfy volume and safety constraints. As a consequence, Navy research includes fundamental detonation studies of liquid fuels and air, which provides a significant challenge because of the difficulty associated with initiating and sustaining detonations with liquid fuel-air propellant combinations[19].

PDE performance estimates and operational range are compared to other candidate propulsion technologies in Fig. 14.

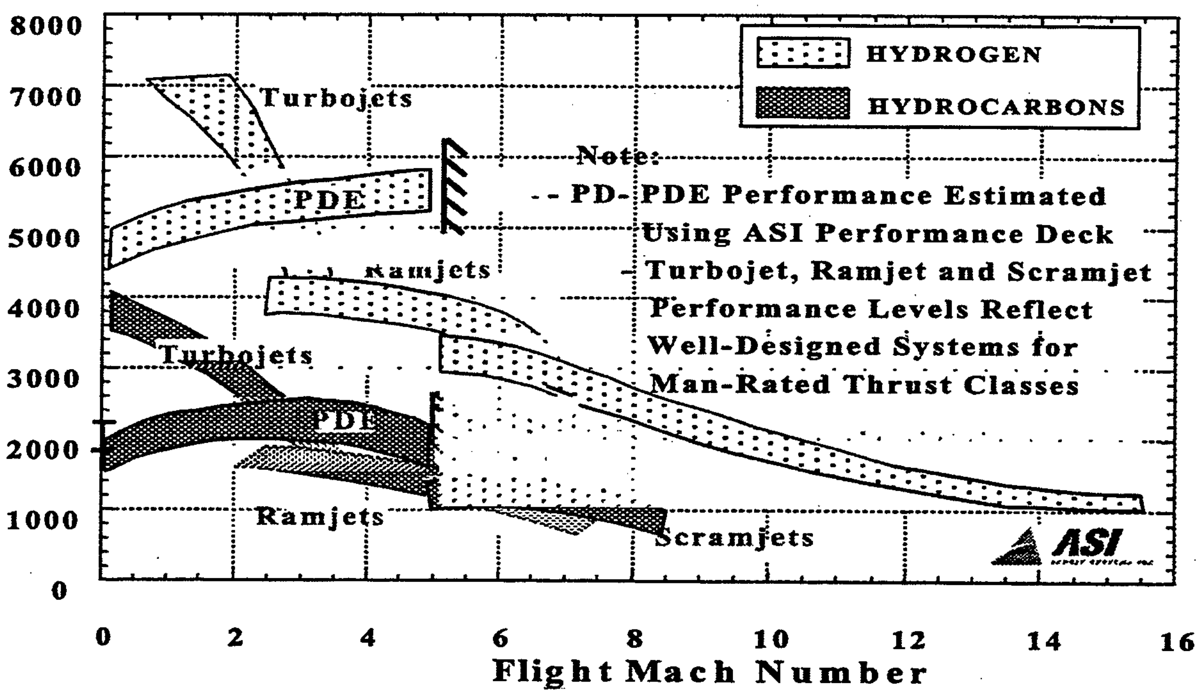

Figure 14. PDE Operational Envelope[50]

The data in Fig. 14 has been provided by Pratt & Whitney Aerosciences, and suggests that airbreathing pulse detonation engines operating with hydrocarbon and hydrogen fuels can outperform other candidate propulsion technologies above Mach 2. The data also suggests that PDEs can provide relatively steady performance over the entire M=0 to 5 operational range.

The Office of Naval Research (ONR) is sponsoring fundamental research and exploratory development of airbreathing pulse detonation engine technology. The ONR exploratory development initiatives include a Multidisciplinary University Research Initiative (MURI) with six participating Universities, international scientific research initiatives, and scientific research initiatives at the Naval Postgraduate School, Naval Air Warfare Center, and Naval Research Laboratory. From these activities ONR has organized an integrated research team to develop an understanding of fundamental detonation science for propulsion applications and requirements for PDE development. The primary objective of the Navy program is to advance the science and

technology sufficiently to develop a viable, more efficient, lower cost alternative propulsion source than is currently available for naval applications[19, 26].

ONR sponsored a series of workshops during the mid to late-1990s to explore the scientific and technical issues associated with PDE technology development, and develop and prioritize scientific approaches to solving them[19, 50]. Based on the findings and recommendations of the workshop participants, ONR initiated the MURI in early 1999 to address the technology development issues identified by the workshop participants. The MURI has been planned as a 3-year plus 2-year follow-on effort[19].

The ONR MURI teams include: 1) Pennsylvania State University Propulsion Engineering Research Center, California Institute of Technology, and Princeton University; and 2) University of California at San Diego, Stanford University, and University of Florida[20]. The University teams are augmented with the support of research staff and test facilities of the Naval Postgraduate School, the Naval Research Laboratory, and the Naval Air Warfare Center[19, 50]. The ONR research effort includes some international participation[19, 50]. Figure 15 outlines the ONR research roadmap and the responsibilities of the participating institutions.

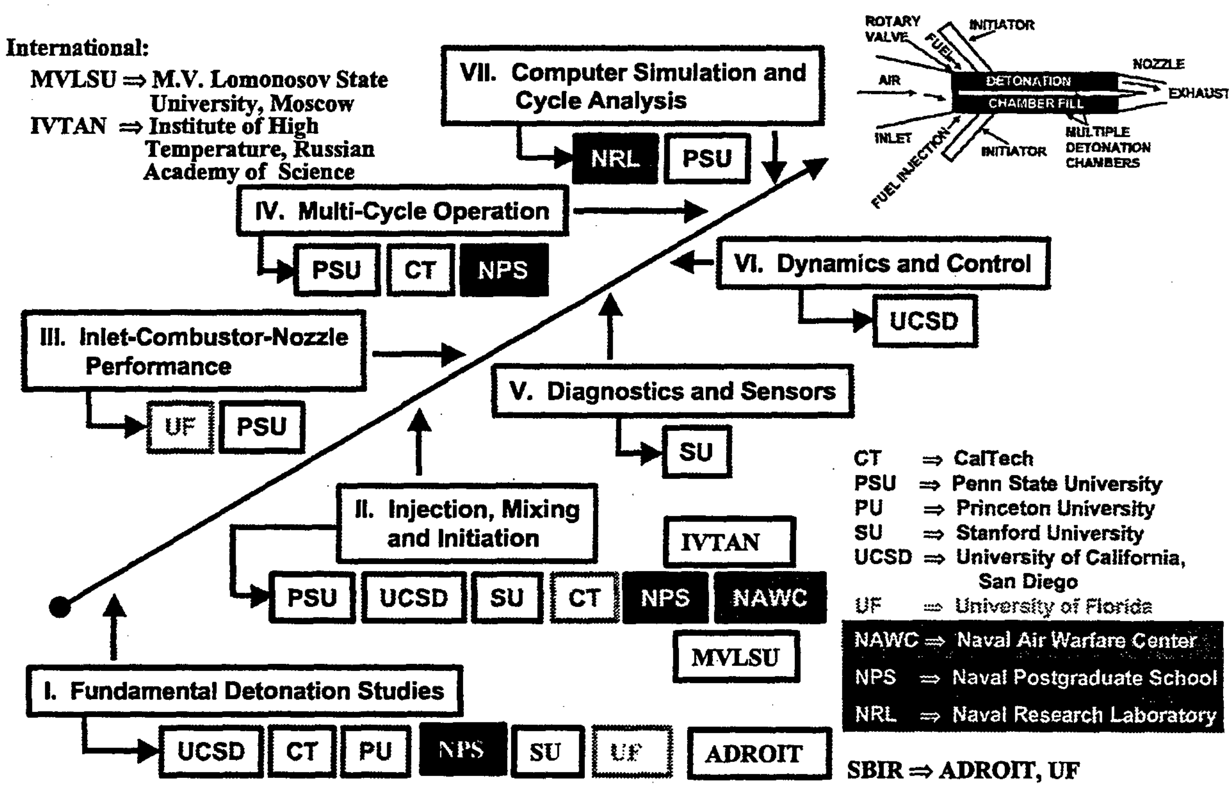

Figure 15. ONR PDE Research Road Map[50]

The ONR research initiatives are intended to advance PDE science and technology and lead directly to exploratory development of a practical PDE. The current research includes fundamental detonation studies with select propellant combinations, investigation of propellant injection, mixing, and detonation initiation characteristics, component design and performance analysis, component

integration studies, diagnostic, sensor, and control requirements, investigation of multi-cycle operation, and advancement of performance prediction and cycle analysis capabilities [50].

Industry has demonstrated multi-tube, multi-cycle operation of laboratory-scale PDE hardware with gaseous propellants [30, 51]. The ONR MURI and core research program seeks to supplement industry's technology advancements with development of a broad scientific database to support further exploratory and advanced development efforts with practical fuels. An ONR-funded research engine has been developed and is in operation at the Naval Postgraduate School to support the Navy MURI and core PDE research program [47]. Detonation experiments have been performed with the ONR research engine to determine the detonability limits of JP-10/air. In addition to the engineering issues regarding valve design, thermal management, and mechanical loading, the ONR research program is investigating numerous scientific issues that must be understood before a practical flightweight engine can be demonstrated [19].

The seven primary areas of research currently being addressed by the combined MURI and core research efforts are shown in Fig. 15. Figure 15 also shows the University teams and government laboratories participating in the various initiatives. There is a great deal of overlap and synergism among the research activities being performed under the various MURI and core research initiatives. Initiative-1 objectives include, but are not limited to, development of a sound understanding of the complex physical, chemical, and thermodynamic phenomena associated with gaseous and liquid phase injection, mixing and ignition, factors that influence rapid development of planar detonation waves, and the role of transverse waves in the detonation process. These theoretical and experimental studies will help to establish minimum ignition energy requirements for practical fuel/air combinations. Recent publications discussing the progress of fundamental detonation studies in support of Initiative-1 objectives are provided in References 52-60.

Initiative-2 research efforts are intended to develop an understanding of propellant injection, mixing, and detonation initiation requirements. Efficient PDE operation will require rapid injection and atomization of relatively large amounts of liquid fuel per detonation cycle. Rapid atomization is required to ensure fast and reliable detonation initiation. Researchers participating in Initiative-2 activities are currently investigating atomization techniques, detonation initiation sensitivity to variations in fuel droplet sizes, local degree of mixing, turbulence, initial propellant mixture temperature, and other parameters. In addition, methods of controlling detonation frequency and deflagration-to-detonation transition times are also under investigation. Recent publications of the ONR MURI initiative-2 efforts are provided in References 61-67. Important objectives of Initiative-2 activities include determination of mixing accuracy control requirements, detonation and DDT control schemes, and minimization of DDT time and distance requirements.

Initiative-3 efforts include investigation of efficient methods of integrating mixed compression supersonic inlets, combustors, and high-performance nozzles. The unsteady behavior of supersonic inlet diffuser flows has been a concern in the development of airbreathing PDEs due to undesirable longitudinal pressure oscillations caused by the cyclic combustion process [68]. The inlet exit plane of multiple-chamber PDEs with a common inlet will experience non-uniform pressure fields arising from operation of the PDE detonation tube valves. The flow area of the inlet/combustor interface changes with time as the air inlet valves for the multiple detonation chambers cycle open and close. Oscillations in backpressure will cause the terminal shock to oscillate about its mean position. Extreme oscillations in backpressures introduce the potential for hammershock and unstarting of the inlet. Therefore, the inlet diffuser must provide a stability margin sufficient for accommodating perturbations of the shock system.

A significant amount of information relevant to initiative-3 objectives is reported in the literature. Researchers have reported extensively on experimental and numerical investigations of diffuser flow with pressure oscillations to understand the effect of combustion instabilities on diffusers as well as the effect of natural oscillations on the combustion chambers[69-78]. These studies have focused

primarily on ramjets with uniform, time-dependent variation of backpressure, and considered only simple normal shocks in convergent-divergent nozzles. Hsieh and Yang have analyzed mixed compression, supersonic inlet flow with consideration of compression processes upstream of the terminal shock [68]. Hsieh and Yang also investigated the response of the shock system to various disturbances, analyzed changes in flow characteristics due to shock oscillations, examined influences of the compression/expansion processes upstream of the terminal normal shock, and examined the influences of the viscous boundary layer and flow separations downstream of the normal shock. Previous theoretical studies have analyzed the flow at the exit of a supercritical inlet of a multiple-chamber PDE and concluded that the time available to transfer air between adjacent tubes while valves are cycling is much less than the time required to form hammershock conditions[35]. These results suggest that the concept of an inlet plenum supplying air to multiple detonation chambers has the potential to become a practical solution for PDE inlets.

The most recent ONR MURI research on this subject is reported in Reference 79. Mullagiri and Segal experimentally simulated the operation of external/internal compression, two-dimensional supersonic inlet by varying the flow area from 32-83% blockage with excitation frequencies ranging from 15-50 Hz. The experimental results indicated that the magnitude of pressure oscillations increased with increasing blockage and decreased with increasing excitation frequency. However, the inlet started and remained started over the entire range of test conditions. These results support earlier analytical studies that shock displacement amplitudes are inversely dependant on backpressure excitation frequency.

Initiative-4 research objectives include characterization of the dynamic coupling between detonation chambers on multiple chamber PDE configurations. A great deal of prior research has focused on fundamental detonation studies and cyclic operation of single-chamber devices. Practical engines may incorporate multiple detonation chambers integrated with a common inlet and nozzle to obtain high aggregate operating frequencies and increase time-averaged thrust. Researchers at Pennsylvania State University, the Naval Postgraduate School, and the California Institute of Technology are currently conducting theoretical and experimental investigation of characteristic engine operating environments for multiple detonation chamber propulsion devices. These efforts will advance the current level of understanding of structural loads, heat transfer characteristics, and performance levels associated with multiple detonation chambers discharging into a common nozzle assembly.

Cyclic discharge from multiple detonation chambers into a convergent nozzle section induces high-amplitude pressure oscillations and high cyclic loading of engine components, possibly exciting the resonance frequencies of the nozzle. The oscillating backpressure will affect overall engine performance and may create high structural fatigue environments. Results of Initiative-4 research will help to establish engine operating environments, structural loading, heat transfer characteristics, cycle-to-cycle performance losses, and chamber purging requirements. Results of ongoing Initiative-4 research efforts will be presented at the Fourteenth ONR Propulsion Meeting in Chicago, IL. in August 2001 and will be included in the ONR meeting proceedings.

Initiative-5 research efforts are focusing on development of practical solutions for obtaining high-resolution experimental measurements for PDE development and validation of computational models. Stanford University researchers are presently developing and demonstrating three diode laser-based diagnostics concepts to obtain in situ measurements of PDE flow properties. The newly developed diode laser-based absorption techniques are used to measure species concentration, temperature, velocity, soot concentration, spray characterization and fuel/oxidizer mixture as a function of location in detonation experiments. ONR research emphasizing standardization of thrust measurement techniques can benefit future efforts to compare performance data obtained from various different PDE experimental test assemblies. In addition, coupling fiber optics with diode laser diagnostics may lead to development of flight-weight spatial and temporal diagnostics and control systems.

Stanford scientists are conducting ignition time studies and measuring individual species concentration time-history profiles in shock tube experiments using the newly developed diagnostic techniques. The experiments are conducted using gas and liquid fuels to support development and validation of chemical kinetics models and demonstrate the new diagnostic methods. Recent results of the Stanford diagnostic concepts development efforts are presented in Reference 80.

Initiative-6 activities are investigating adaptive, active control techniques to ensure optimal PDE performance while maintaining a margin of operational stability. PDE performance optimization requires active control of the cyclical detonation process over a wide range of flight conditions. The science and technology communities are still very uncertain as to what control parameters yield the most effective control of pulsed combustion processes. Ignition timing, valve sequencing, injection, and fuel distribution are some of the candidate parameters for primary means of active combustion control. The University of California San Diego researchers are investigating applicable control theories and feedback requirements. The UCSD research team may present results of their current activities at the Fourteenth ONR Propulsion Meeting in Chicago, IL. in August 2001.

Initiative-7 activities include development and application of computational tools to improve overall understanding of the operation and performance of PDEs. In addition, development and validation of analytical design tools is also underway. The Naval Research Laboratory (NRL) is conducting computational studies of various aspects of PDE operation, including fuel/air mixing, detonation initiation and propagation, multiphase effects, and estimation of idealized PDE Performance [25]. NRL has reviewed past and recent experimental, theoretical, and computational results assessing PDE performance in order to gain insight into reasons for variations in reported results [18, 25, 45, 81, 82]. The key objectives of Initiative-7 are to advance the scientific understanding of PDE inlet-combustor-nozzle interactions, component parameterization, and system design optimization [19].

As part of the ONR MURI and core research programs, the Pennsylvania State University [83] and the Montana State University [84] are developing PDE thermodynamic cycle analysis and design optimization tools. The unsteady nature of the engine operation makes engine evaluation and optimization very difficult. Complete experimental characterization of PDE flowfields is also impossible. Therefore, development of a well understood numerical models is critically important to assist designers and experimenters advance development of practical devices. Design optimization challenges being addressed include predictable, repetitive initiation of detonations, minimization of ignition energy and mixture enrichment, design of responsive, fast acting propellant valves and propellant distribution manifolds, and ignition control [83, 84].

Figure 16 identifies the objectives of each of the ONR-sponsored initiatives shown in Fig. 15. Major objectives of the ONR effort include determination of minimum ignition energies and DDT distances; propellant management (injection, mixing, ignition) and inlet/combustor/nozzle performance over a range of flight speeds, and development of engine control strategies. Data generated from each of the ONR activities will be used to validate and enhance the fidelity of existing simulation tools.

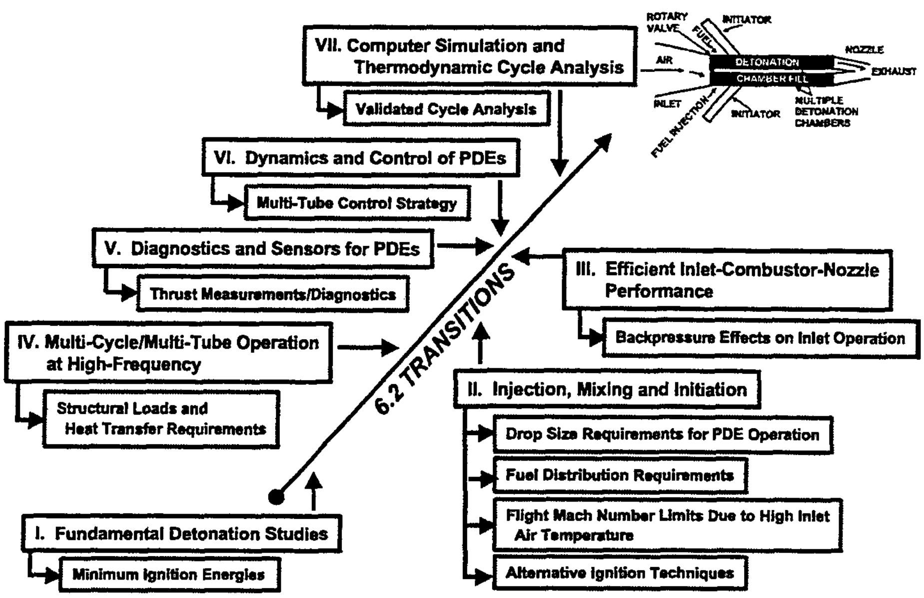

Figure 16. Expected Transitions From ONR PDE Research [R-29]

ONR has also initiated a 4-year, 6.2 risk reduction program focusing on development of a supersonic PDE. The risk reduction contractor team includes Pratt & Whitney, The Boeing Company, Pratt & Whitney Aerosciences, and United Technologies Research Center. The ONR team is currently in the second year of the four-year development program and will provide PDE components to the NASA Glenn Research Center to support ground and flight test research [85]. The ONR core and MURI research programs are being closely coordinated in order to identify and accelerate the technology advancements that contribute to rapid demonstration of practical PDE components, subsystems, and systems.

10.22 AIR FORCE

The Air Force Research Laboratory (AFRL) Propulsion Directorate (PR) is assessing the technical merit of airbreathing and rocket-based pulse detonation technologies for Air Force applications. Potential Air Force applications for PDEs include expendable high-speed missile propulsion systems and high-speed aircraft propulsion systems. Potential PDRE applications include upper stage and orbit transfer propulsion systems, and satellite maneuvering, attitude control, and station keeping propulsion systems. If successfully developed, pulse detonation propulsion technologies might be applied to increase the velocity, operational range, and/or payload capability of long-range missile systems and remotely piloted vehicles, increase orbit insertion and orbit transfer payload mass, and increase the operational lifetime of satellite propulsion systems. Pulse detonation technology might also be applied to improve airbreathing engine compression and afterburning efficiencies, and to develop new hybrid engine cycles.

AFRL/PR, WRIGHT PATTERSON AFB

The AFRL Turbine Engine Division/ Combustion Sciences Branch (PRST) is conducting an in-house computational and experimental program at Wright-Patterson AFB to investigate and develop an airbreathing PDE using kerosene based fuel and air [86, 87, 88]. The Air Force PDE experimental development program was initiated in 1997 and has been divided into three areas: (1) modeling, (2) facilities and instrumentation, and (3) research hardware development and testing [86, 88]. AFRL modeling efforts include investigation of optional detonation initiation schemes, DDT and minimization of transition length, and effect of various PDE design characteristics, such as confinement, obstructions, and wave propagation from small diameter to large diameter chambers [86, 88, 89].

AFRL has developed a Pulsed Combustor/Detonation Engine Research Facility and a four-chamber, government owned research engine to facilitate experimental objectives of the in-house program [86-88]. The capabilities of the research facility and design characteristics of the four-detonation chamber research engine are detailed in References 86-88. The AFRL research engine is capable of operating at an aggregate frequency of 400 Hz (100 Hz per detonation chamber) [89].

The experimental research engine is being used to investigate such critical issues as: detonation initiation and propagation; valving; timing and control; instrumentation and diagnostics; purging, heat transfer, and repetition rate; noise and multi-tube effects; detonation and deflagration to detonation transition modeling; and performance prediction and analysis [86-88]. The AFRL research engine has completed initial testing and evaluation, and preliminary results have been obtained with premixed hydrogen-air to demonstrate proof of concept operation and verify model predictions while avoiding the detonation initiation problems associated with more complex hydrocarbon fuels [86]. The unique capabilities of the AFRL in-house program will help to accelerate development of PDE technology while generating publishable PDE data that can be used to develop and validate PDE performance models [87].

AFRL/EDWARDS AFB

The AFRL Propulsion Sciences and Advanced Concept Division (PRSA) at Edwards AFB is conducting an in-house computational and experimental program to assess the technical merit of the pulse detonation cycle and advance PDRE technology development [90]. The AFRL/PRSA program includes modeling, analysis, and experimental assessment of in-house and proprietary pulse detonation systems. AFRL has reported on the development and application of a constant-volume limit model and reported theoretical performance results [91, 92]. AFRL has also initiated a study on the use of condensed phase fuels in PDRE's, including monopropellants, for pulsed combustion concepts [92, 93].

The AFRL pulsed combustion research assumes achieving high rates of heat release may not require detonation when using reactive monopropellants. The non-detonative cycle eliminates the chamber purge portion of the combustion cycle normally required for the rocket-based pulse detonation engine. Reactants are injected into the combustion chamber during the low-pressure portion of the cycle and ignite spontaneously due to the presence of combustion products remaining as a result of the previous combustion cycle. The chamber pressure rises as the combustion products are produced more rapidly than they exit the nozzle, and thrust is produced as the products expand and accelerate through the nozzle. Once the propellants have reacted and expanded through the nozzle the chamber pressure is reduced to the lowest point in the cycle and fresh propellants are injected to initiate a subsequent combustion cycle [90]. This concept does not depend on chamber resonance or meeting the "Rayleigh condition." AFRL is presently developing a numerical model to determine if this strategy is feasible with monopropellant mixtures (nitromethane/methanol) and bipropellants [91]. The numerical model is a lumped parameter code that includes models for fuel injection, heat release, and blowdown. Experimental facilities and equipment have been developed to support the

pulsed combustor design study. The pulsed combustor chamber and injector are instrumented to characterize the heat release rates of the monopropellant and provide anchoring data for the pulsed combustor design code [93].

AFRL is also preparing a laboratory for studies of detonations of liquid oxygen sprays, having recognized a lack of data for this likely propellant of choice for PDRE operation [93]. Initial tests will be conducted with LOX/GH2. Higher liquid fractions will then be explored using gaseous hydrocarbon fuels [93]. Test results will determine if liquid hydrocarbon/liquid oxygen tests are warranted. Cryogenic liquid methane tests may also be conducted [93].

AFRL has developed a state-of-the-art PDRE test facility at Edwards AFB to experimentally assess the performance of PDRE system designs. This facility has been used to support a joint NASA/AFRL engine demonstration effort being performed by the Pratt & Whitney Aerosciences Center. Accomplishments and planned research of this test program are discussed in more detail in Section 10.3.

10.23 DEFENSE ADVANCED RESEARCH PROJECTS AGENCY (DARPA)

The Defense Advanced Research Projects Agency (DARPA) is developing a new, small-scale class of propulsion systems to enable future development of very small weapons and military platforms [94]. Development of miniature propulsion systems on the order of 0.5-6 inches in diameter and 0.02-20 lbf thrust are envisioned to propel Micro Air Vehicles (MAVs) and Unmanned Combat Air Vehicles (UCAVs) that will be applicable to a variety of existing and new missions. New self-propelled, small scale platforms could compliment emerging unmanned aerial vehicle technologies by extending the range and reducing the radar cross-section of unmanned reconnaissance and surveillance systems, enable covert tagging of high-value targets, and improve the precision of unmanned weapons delivery systems. The DARPA Tactical Technology Office (DARPA/TTO) is currently funding development of miniature propulsion technologies and micro air vehicles through the Small Scale Propulsion Systems (SSPS) and Micro Air Vehicle technology programs. Pulse detonation propulsion technology is one of the leading candidates being investigated to achieve the desired reductions in aerial vehicle dimensions while simultaneously meeting propulsion system performance requirements [94].

DARPA/TTO is funding a joint effort to develop an airbreathing pulse detonation engine that operates with JP-8 fuel on the smallest possible scale. The General Electric (GE)/Lockheed Martin/Stanford University/California Institute of Technology team has been tasked to develop the necessary technologies to demonstrate a pulse detonation engine that develops 20 lbf thrust and meets stringent packaging requirements (~ 12 inches in length). The contractor/university team are investigating the combustion physics, flight dynamics, thermal and mechanical stress environments, required material technologies, and manufacturing techniques that will allow successful development of lightweight, low cost, expendable engine that satisfies all performance requirements [95]. Advanced coatings and refractory materials are being considered for use in the valves and air induction system, and Micro Electro Mechanical (MEMs) technologies are being incorporated for valve/control systems [95].

10.3 NASA RESEARCH CENTERS

NASA GLENN RESEARCH CENTER

NASA is currently exploring the use of pulse detonation combustion for all areas of aeropropulsion, including high performance aircraft, access to space, and commercial aviation. NASA is conducting a coordinated research and development effort of airbreathing PDEs in support of NASA and NASA customer missions, which include commercial and military applications of flight technologies from

subsonic to supersonic and hypersonic regimes [96]. NASA Glenn Research Center (GRC), Langley Research Center (LaRC), and Dryden Flight Research Center (DFRC) are involved in efforts ranging from systems analysis to integrated PDE system flight research. A programmatic roadmap of NASA GRC-led development activities for airbreathing PDE is shown in Fig. 17. Efforts are mapped relative to the potential application and the corresponding operational mode of a PDE-based propulsion system; "pure" PDE for stand-alone PDE combustion chambers, hybrid for combinations with turbomachinery, and combined-cycle when incorporated with a thermodynamic cycle other than the PDE cycle. Pure PDEs are envisioned for expendables and high-performance military vehicles. Hybrid PDE operation is envisioned for supersonic vehicles and commercial applications, and combined-cycle PDE systems are applicable to hypersonic flight and access to space missions [96].

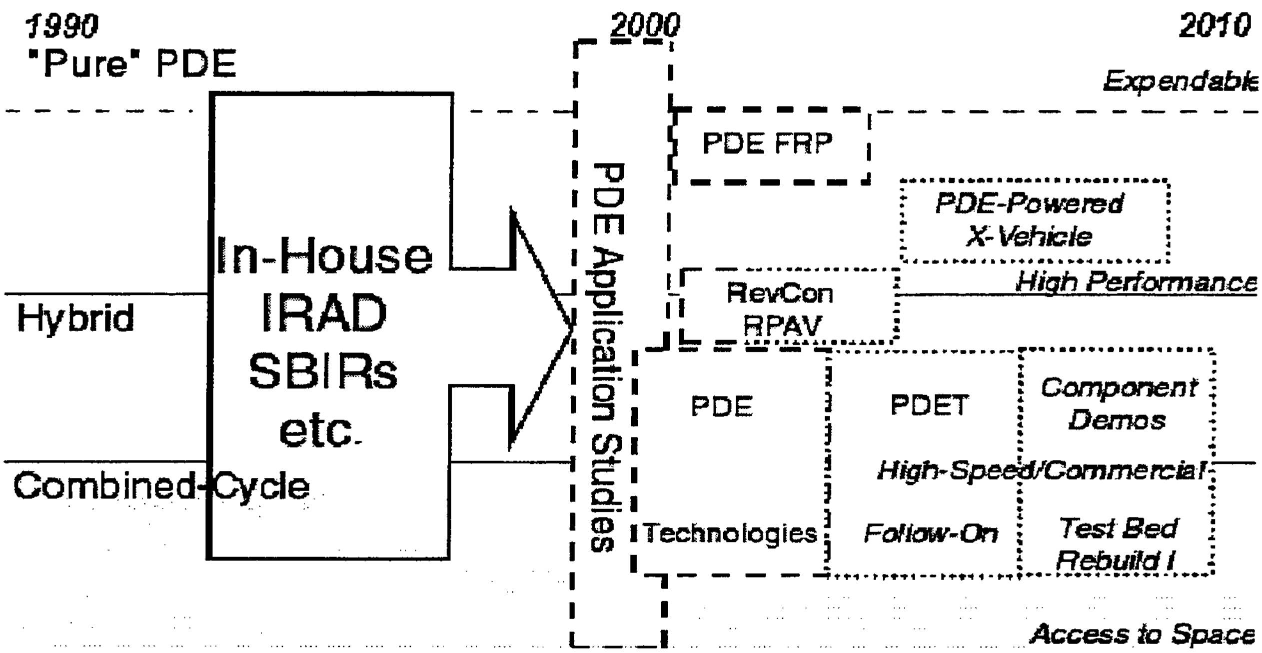

Figure 17. NASA Glenn Research Center PDE Technology Development Roadmap [96]

NASA has initiated direct in-house efforts on PDEs and related technologies, cooperative internal research and development (IRAD) efforts with US Industry, and small business initiative research (SBIR) efforts. It is anticipated that these efforts will continue and increase in investment as PDEs continue to show promise. Current NASA GRC efforts are depicted by the black, bolded, dashed lines. Potential future efforts envisioned are depicted by black dotted lines. Brief descriptions of the current efforts follow.

A comprehensive systems benefits comparison of PDEs relative to existing or competing notional propulsion systems is being conducted by NASA as part of the PDE Application Studies for a multitude of potential flight applications. This systems analysis effort is being conducted by Boeing Phantom Works under a task order contract managed by NASA LaRC. A broad range of potential vehicle class applications for PDEs has been considered, ranging from access to space to Unmanned Aerial Vehicles (UAVs) and rotorcraft.

NASA GRC has initiated a Base Research and Technology effort for PDE development called the Pulse Detonation Engine Technologies (PDET) project. It is primarily a NASA GRC "in-house" conceptual design and technology development effort, but also includes cooperative efforts and/or grants with universities, industry, and other government agencies. PDET is currently planned as an

$11M, 3-year effort initiated in fiscal year 2000, but is expected to have follow on efforts assuming PDEs still show promise. PDET's primary objective is to determine the viability of airbreathing PDE-based propulsion systems for missions of interest to NASA and NASA customers, primarily addressing hybrid and combined-cycle modes of PDE operation. Tasks under this effort include combined-cycle and hybrid conceptual design, cycle analysis, ejectors, inlets, nozzles, acoustics, combustors, materials and structures. Figure 18 is a schematic representing the various tasks and their respective timeframes within PDET. The project is fairly new. Significant technical results are not expected until mid calendar year 2001.

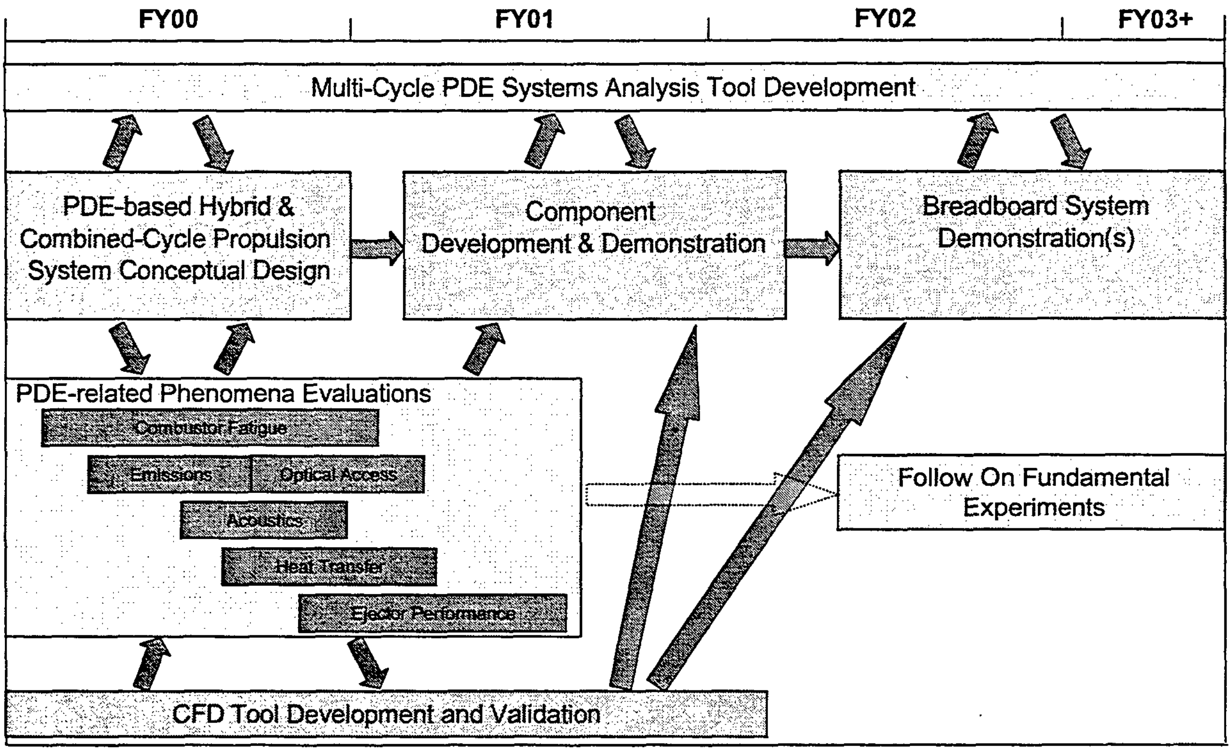

Figure 18 **NASA Glenn Research Center Pulse Detonation Engine Technology Program**

A PDE Flight Research project is also underway that will ultimately conduct ground and flight research tests of an integrated PDE system. It is anticipated that an integrated inlet/combustor/nozzle will be flown on an F-15 supersonic aircraft after completion of successful ground test operations. Boeing Phantom Works is the prime contractor for this teamed effort. NASA GRC, NASA DFRC and NASA LaRC are involved with aspects of the ground tests, flight research tests, code validation, and inlet development. NASA GRC is leading this effort. The total program resources are approximately $11M over three years. The PDE Flight Research project is funded entirely by NASA's Revolutionary Concepts in Aeronautics (RevCon) program.

The goals and objectives of the RevCon program focus on the use of flight to advance aerospace technologies in an accelerated fashion. The primary objective of the PDE flight research program is to raise the Technology Readiness Level (TRL) of an integrated PDE system for near-term application. To meet this objective, the flight research team must develop state-of-the-art PDE system technologies, and identify and resolve technological problems through system integration, ground, and flight test operations.

The primary technologies that will be developed in the PDE Flight Research Project include an unsteady inlet, engine system integration, and code validation using results from wind tunnel and flight research tests. NASA is coordinating this effort with the Office of Naval Research (ONR) PDE Risk Reduction Program in that the engine hardware being developed under that effort will be used for the Flight Research Project [96]. The flight research project team will develop and demonstrate inlet technologies, integrate PDE subsystems (inlet/combustor/nozzle) within a flight test fixture for wind tunnel and flight test operations, conduct wind tunnel tests to acquire integrated PDE performance and operability data, and conduct flight research tests to validate PDE system performance and operational envelope [96].

The RevCon program is also funding the Revolutionary Propulsion for Aeronautical Vehicles (RPAV) effort. A General Electric Aircraft Engines (GEAE)-led team was recently awarded a contract to determine the feasibility of using a PDE as an augmentor. Internal studies conducted by GEAE and their partners have indicated potential improvements in thrust and efficiency from using a PDE augmentor on a typical military fighter engine. The scheduled completion date for the Phase 1 contract effort is June 2001. A Phase 2 effort is possible following a competitive process within the RevCon program [96].

NASA LANGLEY RESEARCH CENTER

The NASA Langley Research Center (LaRC) is presently sponsoring the PDE Applications Study discussed above to help facilitate interagency PDE program planning and development of technology investment strategies [96, 97]. LaRC has developed high-speed fuel valves for PDE applications, and has recently completed design and performance analysis of supersonic PDE air induction systems with industry partners Pratt & Whitney Aerosciences Center and Lockheed Martin Tactical Aircraft Systems (LMTAS) [35, 98]. This effort included conceptual design and integration of an integrated PDE propulsion system with a supersonic aircraft; time-dependent CFD analysis of the inlet flowfield, estimation of installed PDE cycle performance over the M = 1 to 3 flight regime, and assessment of high Mach number performance of the integrated inlet-PDE system [35].

LaRC has also completed a Small Business Innovative Research (SBIR) contract with Pratt & Whitney Aerosciences Center to assess the performance of a turbofan with PDE duct burning for Earth-to-orbit vehicle applications [98]. The PDE duct-burning concept uses PDEs in the fan flow. The SBIR contract included a comparison of the PDE duct-burning turbofan performance with conventional afterburning turbojet performance. The goal of the NASA LaRC efforts includes development of a highly efficient propulsion system to power space launch vehicles through the low-speed (M = 0 to 3) portion of the flight trajectory [35, 98].

NASA MARSHALL SPACE FLIGHT CENTER

The NASA Marshall Space Flight Center (MSFC) is sponsoring development of PDRE technology as part of the Advanced Space Transportation Program (ASTP) [44]. MSFC is conducting in-house research and modeling activity [99, 100], and has also awarded separate contracts to Pratt & Whitney Aerosciences Center and United Technologies Research Center (UTRC) to develop and demonstrate unique PDRE concepts [44].

The MSFC in-house activity includes development and operation of a single chamber PDRE and model development activity. MSFC has developed gaseous H_2/O_2 rocket engine simulator in the Advanced Propulsion Research Laboratory to meet a number of objectives. The in-house development effort included design and fabrication of electronic control circuits, a spark ignition system, coaxial injector, initiator tube, detonation initiator, and combustion chamber. Data collected from low-frequency test firings will be used to support development of theoretical/CFD

analysis tools and improve definition of system operational requirements. The engine simulator will also be used as a test bed for major subsystem and component designs to explore alternative engine design configurations, and optimization of propellant injection techniques and nozzle designs. In addition, the simulator tests will improve understanding of detonation physics, help to validate PDRE scaling laws, and lay the groundwork for liquid propellant PDRE operation. MSFC is also conducting an in-house PDRE injector design and development activity that is directly associated with the in-house engine simulator activity [99, 100].

MSFC is sponsoring the University of Tullahoma Space Institute (UTSI) PDRE model development activity to establish a government-owned, non-proprietary PDRE performance model. The 1-dimensional, CFD, ideal gas model will be used to provide an understanding of shock behavior, reflections, flowfield, variable mixture ratio, and cycle time history. Current activity includes development of multiple chamber/common nozzle modeling capability and incorporation of more general gas properties [101].

The MSFC in-house activities are planned to continue into FY02 to demonstrate PDRE weight and cost goals, complete and release the government-owned performance code, and develop all of the technologies required to enable a flight weight demonstration of PDRE technology.

Pratt & Whitney Aerosciences Center and UTRC are presently in the third year of their original three-year program to develop PDRE demonstration test articles for MSFC and have reported a number of successes. Pratt & Whitney Aerosciences Center has developed a multiple chamber PDRE concept in which the six combustion chambers exhaust through a common nozzle flowpath, thereby providing back pressure for the purge and fill processes. Work on this concept originated in 1992 through an Air Force Research Laboratory contract with Pratt & Whitney Aerosciences Center, then Adroit Systems Incorporated. The objective of the 1992 AFRL program was to demonstrate reliable, repetitive detonation combustion with gaseous oxygen-hydrogen propellants in a single-chamber, proof-of-concept test set-up [102]. Continued efforts with Air Force and internal Pratt & Whitney Aerosciences Center funding led to additional development and demonstration test experience and eventually development of a first generation, multi-chamber, laboratory-scale experimental engine [102].

Pratt & Whitney Aerosciences Center initiated numerical and experimental investigation of this flowpath concept in 1997, and has demonstrated operation of the water-cooled test article with gaseous hydrogen/gaseous oxygen propellants at ambient and elevated fill pressures and a range of propellant mixture ratios [30]. NASA MSFC elected to leverage this technology development effort and contributed additional funding to continue testing with a larger test article [102]. The NASA MSFC/AFRL/Pratt & Whitney Aerosciences Center team has demonstrated combustor-firing frequencies of 80 Hz, corresponding to an aggregate 480 Hz engine frequency for short durations, and has also demonstrated altitude start capability (200 kft). Pratt & Whitney Aerosciences Center is also developing high-speed propellant valves and conducting experimental investigation of PDRE injector head configurations to improve atomization, mixing, and transient response [30]. The objective of the MSFC/Air Force/Pratt & Whitney Aerosciences Center team is to develop a practical system that will demonstrate and validate high performance predictions and high reliability with low-complexity, low-cost, and lightweight design characteristics. The current focus of the joint MSFC/AFRL effort is to demonstrate significantly improved performance of the P & W Aerosciences PDRE through engine flowpath refinements and collect comprehensive data sets that can be used for anchoring modeling and simulation tools and standards for reliable prediction of PDRE performance [102].

UTRC is teamed with Advanced Projects Research Incorporated (APRI) to develop a single chamber, H_2/O_2 PDRE for NASA MSFC. The UTRC team engine incorporates an aerodynamic throat to provide the necessary engine backpressure. The UTRC team has demonstrated operation of their PDRE concept with a mechanical throat, and is planning to initiate system level testing with the

aerodynamic throat configuration in 2001. UTRC is also developing unique propellant valves for their PDRE concept that will be incorporated in the engine demonstration tests beginning in 2001 [44, 103].

11.0 PRACTICAL ENGINEERING ISSUES

Significant advancements have been made in pulse detonation propulsion science and technology over the past several decades. Nichols, et al, established an analytical theory for pulsed combustor performance with favorable comparisons with experimental data [104]. Schmuel Eidelman of SAIC and colleagues at the Naval Postgraduate School developed and demonstrated operation of self-aspirating pulse detonation engine in the mid-1980s [22]. The SAIC/NPS test article is considered one of the first modern PDE concepts and was evaluated in a number of analytical and experimental studies [105]. Since that time, fundamental detonation studies and component development and test programs have reported measurable success. In addition, simulation tools and cycle analysis capability have advanced and have significantly aided component design and integration capabilities. However, many model development and system design challenges remain that are currently being addressed.

The ability to reliably initiate and sustain detonations over a range of operational conditions with practical fuels remains to be demonstrated. Optimum inlet and inlet/detonation chamber interface designs for airbreathing PDEs introduce significant engineering challenges. Engineering design solutions that minimize external flowfield losses and off-nominal performance losses are yet to be developed. Propellant manifold/mixing systems must be designed to reliably combine and deliver uniform, detonable fuel/air (PFE) and fuel/oxidizer (PDRE) mixtures to the detonation chambers over the expected range of flight conditions to ensure continuous operation. Reliably establishing and controlling detonations with practical fuels has proven to be extremely challenging because the detonation processes are very sensitive to stoichiometry, particle/droplet size, local degree of mixing, etc. [32]. Flight weight high-speed propellant valves and control system components require additional development effort and will be subject to unique system design constraints. Detonation chamber purging and refilling must be reliably repeated on very short timescales to ensure against premature ignition of fresh propellant charges. The pulsing mode of PDE and PDRE combustors introduce nozzle integration challenges to ensure nozzles flow full.

Integration of PDEs with turbomachinery designs may also introduce significant design challenges. Turbines are typically designed for steady state, homogenous flow. In addition, many applications that might incorporate a hybrid PDE are noise sensitive. A hybrid PDE might be able to take advantage of bypass air, turbomachinery, and possibly active cancellation for acoustic suppression. However, an understanding of the acoustic signature of an installed PDE device is not currently available [106].

The above comments summarize a few operational design challenges that have not been completely addressed, and design solutions for each of these operational issues will likely be system specific. In addition, vibration abatement, noise abatement, heat transfer, structural loading and cycle fatigue characteristics will need to be addressed for each application.

Government efforts to coordinate development of performance codes are intended to standardize methods of assessing performance that will accurately predict PDE/PDRE performance benefits. The unsteady nature of pulse detonation propulsion devices introduce new system modeling challenges. PDE design tools need to accurately model inlet operation. PDE and PDRE design tools need to model fuel injection, fuel-oxidant mixing, ignition, detonation and wave propagation, nozzle flow, expansion of combustion products, and purging of the combustion chambers. Evolution of standardized performance modeling techniques is envisioned to require a 2 to 3 year effort. In order to understand the performance of individual components, component models must be developed and validated. Once developed, component models for inlets, feed system valves,

combustors, nozzles, etc., can be integrated into systems level codes that include appropriate chemistry models. Validated system level codes can then be used to establish system level benefits in terms of specific fuel consumption, system weight, development cost, and life-cycle cost. In addition, system level performance predictions will account for real-engine effects and losses due to component efficiency limits.

PDE and PDRE component and subsystem development efforts can be conducted in parallel with model development activities to generate test data for validation purposes. Mature component and subsystem development efforts could then lead to a flightweight engine test program to demonstrate actual performance levels attainable with pulse detonation propulsion devices.

Conventional airbreathing engines make use of accessory drives and other devices to tap-off energy from rotating engine components in order to generate electrical power, pressurize hydraulic systems, and pressurize cabins. The simplicity of pulse detonation propulsion devices severely limits power extraction by conventional means. Development of innovative techniques and technologies may be necessary to incorporate power extraction from PDE/PDRE systems. Although power extraction is beyond the scope of present-day pulse detonation technology initiatives, power extraction will ultimately need to be addressed or replaced with alternate sources of power for operation of aircraft subsystems.

12.0 SUMMARY

Pulse detonation propulsion technology may become an attractive option for missile, air, and space transportation systems in the 21st century if current scientific and engineering obstacles are successfully addressed. Because detonation is an extremely efficient means of burning propellant mixtures to release the chemical energy content, pulse detonation propulsion technology is anticipated to yield a greater performance payoff than other competing technologies, such as combustion control or advanced fuel development. The higher efficiency of the detonation cycle introduces the potential for significant improvement in overall system performance with simultaneous reduction in weight, complexity, cost, and packaging volume requirements, relative to conventional aircraft and space propulsion systems.

The DoD, NASA, and commercial enterprises are pursuing fundamental research and exploratory development initiatives to understand the fundamental physics involved with initiating, sustaining, and controlling detonations, and to develop and demonstrate airbreathing, hybrid, and rocket-based propulsion system designs. Primary applications of pulse detonation propulsion technology include stand-off munitions, precision-guided munitions, tactical and long range missile systems, military and commercial air transport, supersonic fighter aircraft, hypersonic vehicles, launch vehicle upper stages, and space propulsion. Other applications are also receiving consideration.

Pulse detonation propulsion science and technology is presently at an early stage of development but is maturing rapidly. Results of experimental detonation studies and combustor test data are being used to enhance and validate numerical models. Experimental PDE and PDRE test articles have been demonstrated at the component and subsystem level, and combined cycle propulsion systems are being studied extensively. DoD and industry initiatives are on schedule to complete significant demonstration tests and transition many new technologies to advanced stages of development in the next few years.

13.0 ACKNOWLEDGEMENTS

The author wishes to express his appreciation to Dr. Chris Brophy of the Naval Postgraduate School and Mr. Ron Fry of CPIA for their assistance and review of a preliminary version of this document. In addition, the author wishes to thank Mr. Robert Pegg of the NASA Langley Research Center, Mr. Joe Doychak of the NASA Glenn Research Center, Dr. Ed Coy of the Air Force Research Laboratory, and Dr. Tom Bussings of Pratt & Whitney Aerosciences Center for their review of a preliminary version of this technology review.

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- OTHER GOV'T. –

DEF TECH INFO CTR/FT BELVOIR
- 2 DTIC-OCC

DTRA/ARLINGTON
- 1 FRANK TRACESKI

- NON-GOV'T. -

AEROJET/SACRAMENTO
- 1 AEROJET-TIC

AEROSPACE/LOS ANGELES
- 1 BENITA CAMPBELL, M1-199

ALLIANT AEROSPACE COMPANY/MAGNA
- 1 LIBRARY, M/S H

ALLIANT MISSILE PRODUCTS COMPANY LLC/ROCKET
- 1 DOTTIE LYON

ATLANTIC RESEARCH CORP/GAINESVILLE
- 1 TECHNICAL INF CTR

BOEING COMPANY/CANOGA PARK
- 1 H. E. SNELL, TIC BA29

BOEING/SEATTLE
- 1 LIB ACQ

LLNL/LIVERMORE
- 1 BETTE MOORE

PACIFIC SCIENTIFIC ENERGETIC MATERIALS CO/CHANDLER
- 1 SMALLWOOD/KORCSMAROS

PENNSYLVANIA STATE UNIV/STATE COLLEGE
- 1 DR. DANIEL KIELY

PRIMEX AEROSPACE COMPANY/REDMOND
- 1 JAMES GURLEY

RAYTHEON COMPANY/TUCSON
- 1 SHANNON MACK

SANDIA NATIONAL LABS/ALBUQUERQUE
- 1 M. GRUBELICH
- 1 S. LANDENBERGER

TALLEY/MESA
- 1 SECURITY OFFICE

THIOKOL PROPULSION/ELKTON
- 1 THOMAS HOLMAN

THIOKOL/BRIGHAM CITY
- 1 ELLEN WAGSTAFF

UNITED TECHNOLOGIES/SAN JOSE
- 1 TECHNICAL LIBRARY

UNIVERSAL PROPULSION COMPANY INC/PHOENIX
- 1 JAMES J. BAKER

Total No.: 67